NACA

RESEARCH MEMORANDUM

CORRELATION OF WIND-TUNNEL AND FLIGHT DETERMINATIONS

OF THE BUFFET SPEED OF AN AIRPLANE EQUIPPED

WITH EXTERNAL STORES

By

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
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SUMMARY

Tests were made in the Langley 7- by 10-foot tunnel on a model of a fighter-type airplane with external stores. This paper is concerned mainly with presenting the data obtained in this investigation and with a comparison of some of these data with flight-test results to determine the feasibility of estimating flight buffet Mach number from tunnel data.

The results of this investigation indicate that the incremental drag coefficient due to external stores may be used to estimate the maximum Mach number that the airplane may reach in flight when it is equipped with external stores. This estimation is conservative for the five configurations investigated by amounts varying from 0 to 10 percent of the flight limit Mach number. The free-stream tunnel Mach number corresponding to sonic flow over the lower surface of the wing in the region of the store is a good indication of the lower limit of buffet in flight of the airplane when equipped with external stores. The fluctuations of total pressure over the horizontal tail are not sufficiently large (maximum of 1 percent q_o) to cause buffeting of the airplane.

INTRODUCTION

It has become standard practice on military airplanes to attach additional armament and fuel tanks externally to the aircraft to increase its utility and to facilitate maintenance. When airplanes equipped with external stores reach relatively high flight speeds a phenomenon is encountered that impairs the efficiency of the airplane as a gun platform and restricts its maximum speed. This phenomenon,

called buffeting, has been characterized as a progressively increasing
vibration of the airplane with increase in flight speed.

The effect of external stores on the aerodynamic characteristics
of airplanes is being investigated by the National Advisory Committee
for Aeronautics. The first part of this program was the deter-
mination of the low-speed aerodynamic effects of external stores on
the stability of models of military airplanes (reference 1). A
theoretical approach to the problem was also published and is presented
as reference 2. Flight tests were conducted on an airplane equipped
with external stores. The flight results are confined to the Mach
number at which buffeting of the airplane becomes objectionable in
the opinion of the pilot and are summarized in table I.

The present investigation was conducted in the Langley high-
speed 7- by 10-foot wind-tunnel to determine the feasibility of
using wind-tunnel data as a means of predicting flight buffet Mach
number. No attempt is made in this report to present a detailed
analysis of the phenomenon of buffeting. This paper is confined to
a presentation of the data obtained by this investigation and a
comparison of wind-tunnel and flight-test results.

COEFFICIENTS AND SYMBOLS

The coefficients and symbols referred to in this report are
defined as follows:

ΔC_D change in drag coefficient due to external store, positive
when store increases the drag $\left(\dfrac{D_{ext} - D_{clean}}{qS} \right)$

ΔC_L change in lift coefficient due to external store, positive
when store increases the lift $\left(\dfrac{L_{ext} - L_{clean}}{qS} \right)$

ΔC_m change in pitching-moment coefficient due to external store,
positive when store increases the pitching moment
$\left(\dfrac{M_{ext} - M_{clean}}{qS\bar{c}} \right)$

C   refers to Mach number or coefficient

D   drag, pounds

L   lift, pounds

M   pitching moment, foot-pounds

$\dfrac{\Delta p}{q}$ pressure coefficient $\left(\dfrac{p - p_0}{q}\right)$

M   Mach number $\left(\dfrac{V}{a}\right)$

q   dynamic pressure, pounds per square foot, $\left(\dfrac{1}{2}\rho V_0{}^2\right)$

ρ static pressure, pounds per square foot

S   wing area (9.85 sq ft on model)

$\bar{c}$ mean aerodynamic chord (1.49 ft on model)

c   chord of wing

V   velocity of air, feet per second

a   velocity of sound, feet per second

α angle of attack of the thrust line

$\beta = \sqrt{1 - M^2}$

ϵ blockage correction at high speeds

Subscripts:

o   denotes free-stream conditions

cr   critical — M_0 at which compression shock is formed as mani-
fested by pressure-distribution data

c_D when used with M — denotes M_0 for ΔC_D divergence

cor.   corrected Mach number or coefficient of force or moment

m   measured Mach number or coefficient of force or moment

Note: Mach numbers and pressures without subscripts denote local
conditions

MODEL AND APPARATUS

The Langley high-speed 7- by 10-foot tunnel in which this investigation was conducted has a closed throat, rectangular cross section, and is of the single return type. The degree of turbulence in the air stream is very low. The model used in this investigation was a small scale model of a fighter-type airplane on which extensive external store tests have been made. The model consisted of a fuselage and a wing of NACA 23018 section at the fuselage center line and an NACA 23009 section at the tip. A three-view drawing of the model showing the method of mounting in the tunnel and a typical store installation is shown in figure 1. Photographs of the model mounted in the tunnel are shown in figures 2(a) and 2(b). The locations of the static-pressure orifices on the lower surface of the wing are given in figure 3. The models of external stores used in this investigation were a Lockheed drop tank and a Navy Universal drop tank. Drawings of these stores showing pertinent geometric characteristics and static-pressure orifice locations are presented in figure 4. These stores were attached to the model by small scale duplications of full-scale pylon fairings. Models of these pylon fairings with the locations of pressure orifices are shown in figures 5 and 6. The various combinations of tanks and pylons tested are given in columns 1 through 4 of table I.

The duct configuration of the model and inlet velocity ratio corresponded approximately to that required for high-speed flight condition of the airplane.

Static-pressure measurements were photographically recorded on a 56-tube direct-reading manometer.

The survey of the fluctuations in the total pressure in the wake was made with a standard pitot-static tube, the static- and total-pressure chambers of which were connected to either side of an electrically sensitive diaphragm. This instrument was mounted on a remotely controlled carriage located in the tunnel test section. (See fig. 2.) Fluctuations in total pressure were recorded by movement of a light beam on a photographically sensitive drum of paper rotating at a known speed, thus making it possible to determine the period and amplitude of the wake fluctuations.

TESTS

Procedure

Symmetrical store installations were tested in addition to asymmetrical arrangements because of the mutual interference effects

indicated by flight buffet Mach numbers. It was originally intended
to obtain information on the aerodynamic characteristics of the
model in the presence of 1000-pound bombs symmetrically and
asymmetrically disposed from the plane of symmetry in addition to
the two types of fuel tanks tested. Due to the time available for
this investigation it was necessary to eliminate this part of
the program and also to measure pressures on only a few of the
configurations on which force data was taken. Pressure tubes were
connected for only two "basic" configurations, that is, the Lockheed
tank on a Standard pylon and the Universal tank on a Standard pylon.
By so doing, it was possible to obtain the necessary pressure
measurements for various other store configurations without connecting
additional pressure tubes by the simple expedient of adding either
a store configuration similar to one of the basic configurations to
the opposite wing or by inserting an extension to the pylon fairing.
Due to the expeditious nature of this method of obtaining pressure-
distribution data it was possible to obtain considerably more infor-
mation in the allotted time than would have been possible if configu-
rations requiring additional pressure connections had been investi-
gated.

Static-pressure distributions for all configurations tested were
taken at a constant angle of attack of -2.75° through a Mach number
range extending from 0.3 to approximately 0.715. The results of
these tests are presented in figures 7 to 12. The angle of attack
was selected as that for which the model lift coefficient was com-
parable to the high-speed lift coefficient of the airplane. For
several configurations, additional angles of attack were tested to
obtain an estimate of the general effect of angle of attack upon
the pressure-distribution characteristics. The static-pressure
orifices in the pylons were located very close to the juncture of
pylon and wing (fig. 5) and approximately in line with the static-
pressure orifices of station $5\frac{1}{4}$ on the lower surface of the wing.
The results of the chordwise distribution presented herein at
station $5\frac{1}{4}$ are composed of data obtained over the lower surface of
the wing and over the pylon at the pylon wing juncture. Unfor-
tunately, it was not possible to use some of the pylon orifices
because of the condition of the tubes. The peak minimum pressure
in such cases was based upon an estimated fairing of the data.
Pressure measurements for each Mach number were recorded simul-
taneously at all stations.

Force measurements were made at constant angles of attack of
-3.5°, -2.75°, -1.5°, and 0.25° through a Mach number range similar
to that of the pressure-distribution tests and are presented in
figures 13 and 14. Force measurements consisted of six component

automatically recorded balance readings. Results of these tests are
presented as increments of lift, drag, and pitching-moment coeffi-
cients. These increments were obtained by a subtraction of the data
for the model without stores from the data obtained with stores in
place. Although all forces and moments were recorded, only those
which showed any appreciable difference (drag, lift, and pitching
moment) due to stores are presented.

Wake surveys were conducted in a plane located at the hinge
line of the horizontal tail and perpendicular to the plane of
symmetry. These tests were made at a constant angle of attack
of -2.75° through a Mach number range from 0.3 to approximately 0.650.
The region surveyed extended from the plane of symmetry outboard
12 inches and from the center line of the tank vertically upward 18 inches.

The results of these tests (pressure and force measurements
and wake surveys) were compared with the results of flight tests
which consisted of the Mach number at which buffet was first
encountered (column 5, table I) and the Mach number at which buffet
became objectionable (column 6, table I).

Corrections

All Mach numbers and force and moment coefficients were corrected
for blocking by the following equation obtained from reference 3.

$$C_{cor} = C_m\left[1 - \epsilon(2 - M_o{}^2)\right]$$

where

$$\epsilon = \epsilon_{solid} + \epsilon_{wake}$$

$$\epsilon_{solid} = \frac{0.00955}{\beta^4}$$

$$\epsilon_{wake} = \frac{0.0533\ C_D}{\beta^2}$$

An increment in drag coefficient of 0.0013 has been added to
account for the horizontal buoyancy resulting from the longitudinal
pressure gradient in the tunnel.

DISCUSSION

Pressure Distribution

Pressure distributions over the lower surface of the wing obtained by pressure orifices located on the lower surface of the wing and in the pylon at the wing-pylon juncture are shown in figure 7. As previously noted, pressure could not be transmitted through certain tubes. Unfortunately the orifices for these tubes were located in what appears to be a region of high velocities. However, consideration of the characteristics of the data available indicate that the minimum pressures presented may give a close approximation to the exact minimum pressures.

It has been established that when a local Mach number of one is exceeded on a wing the boundary layer tends to separate, resulting in wake vorticity. The precise local Mach number at which this phenomenon occurs depends to some extent upon initial boundary-layer conditions. It is felt that buffet is first experienced in flight when the turbulent wake associated with local supersonic velocities over the lower surface of the wing passes over the tail surfaces of the airplane. It follows that M_0 corresponding to local sonic flow over the lower surface of the wing might be used as a rough estimate of the flight Mach number at which buffet is first encountered. Pressure-distribution data indicate that the local velocity of sound is first attained on the lower surface of the wing at approximately 28 percent chord. The free-stream Mach number corresponding to this local sonic velocity as determined by cross plotting peak minimum pressure against Mach number and critical pressure against test Mach number is tabulated in column 7 of table I. The agreement between these data and the flight initial buffet Mach number (column 5) is better than might be expected in light of the previous discussion and since the precise attitude of the airplane during the flight tests was unknown. Recent work (unpublished) has indicated that sideslip may have considerable effect upon data of this nature. In addition flight results presented herein were based upon human reaction to the vibrations of the airplane when stores were added and as such should be considered as only a qualitative index to the onset of this disturbance. In column 6 is presented the flight buffet Mach number at which buffeting of the airplane becomes so severe that the airplane was rendered useless as a gun platform or was damaged structurally due to the vibration. Tabulated in column 8 is the free-stream Mach number at which a well defined compression shock is formed on the lower surface of the wing as manifested by a breaking down in the variation of peak minimum pressure against M_0.

It is obvious that this Mach number is, with the exception of one configuration, considerably higher than either the initial buffet (column 5) or the limit buffet (column 6) Mach numbers. It appears, therefore, that the formation of a compression shock may not be essential to the phenomenon of buffeting, but rather that the turbulence associated with local sonic velocities prior to the development of a well defined compression shock is sufficient in itself to be prohibitive.

Pressure distributions over the tanks used in this investigation are presented in figure 8. The velocities over the tanks in all cases are lower than those over the lower wing surface at the same M_o, and, therefore, the critical Mach number of the tanks is higher than the critical Mach number of the lower surface of the wing.

A variation in angle of attack of -3.5° to -1.5° seems to have small effect upon the results discussed previously. Increasing the angle of attack appears, however, to increase slightly the free-stream Mach number that corresponds to the attainment of the local speed of sound over the pylon. This may be attributed to the decreases in the peak minimum pressures of the pylons that are caused by changes in local pressures over the lower surface of the wing.

Force Tests

The results of the force tests are presented as incremental variations due to stores of drag, lift, and pitching-moment coefficients. Results are presented for each tank in combination with various pylons at model angles of attack of -3.5°, -2.75°, -1.5°, and 0.25°. These tests were conducted in an attempt to recognize as discontinuities in the incremental force data those adverse flow characteristics that were shown to exist by pressure-distribution methods. Due to the nature of the problem the variation of ΔC_D with free-stream Mach number was thought to be of fundamental interest. The free-stream Mach numbers for which rapid positive increases in incremental drag coefficient occurred are given in column 9, table I, for each model configuration at an angle of attack of -2.75°. These Mach numbers were obtained from inspection of the curves of the variation of ΔC_D with Mach number. The Mach number for divergence of ΔC_D agrees closely with the limit flight buffet Mach number (column 6). The tunnel results were in all instances either equal to or lower than limit flight buffet Mach number. In view of the qualitative nature of the flight results and the unknown effects of boundary layer and sideslip, it is considered significant that the incremental drag measurements agree within 10 percent with the five flight configurations on which

data were available. It is important to remember, however, that an underestimation of the flight limit buffet Mach number by 10 percent may be of the same order of magnitude as the decrease in the critical speed of the airplane caused by the addition of stores. As might be expected, increasing the angle of attack generally seems to increase slightly M_{c_D}.

It is apparent that the addition of stores to this model generally resulted in a decrease in lift coefficient at constant angle of attack at low Mach numbers. As Mach number is increased the incremental lift coefficient seems to increase positively in an orderly manner for the lower angles of attack. However, for angles of attack of -1.50° and 0.25° the variation of ΔC_L with Mach number does not follow any general pattern. The addition of stores to the model generally caused a positive incremental change in pitching-moment coefficient which increased in magnitude with an increase in Mach number for angles of attack of -3.5° and -2.75°. When the angle of attack was increased to -1.5° the incremental pitching-moment coefficient remained fairly constant to a Mach number of approximately 0.675. Further increases in M_0 resulted in decreases in ΔC_m. At an angle of attack of 0.25° the variation of ΔC_m with M_0 seemed to be rather erratic, although a small general decrease was noted.

Wake Survey

At the onset of this investigation it was felt that if pulsating wake characteristics existed in the plane of the horizontal tail the concurrent changes in aerodynamic characteristics of the horizontal tail (primarily lift) would furnish ample cause for buffeting of the airplane. Such results could be affected by either changes in dynamic pressure or angularity of the wake or a combination of the two. It was, therefore, decided that a survey should be conducted to establish grounds for such a hypothesis. Due to the time limitations of this investigation, it was possible to obtain a survey of the fluctuations of only the total pressure which is analogous to the fluctuations in dynamic pressure.

The results of the survey of the fluctuations of total pressure in the wake are not presented herein because the magnitude of these disturbances over the horizontal tail preclude their analysis as related to buffeting. The maximum fluctuation of total pressure as indicated by this survey was of the order of 6 percent free-stream dynamic pressure and was located on the center line of the tank. However, the amplitude of these fluctuations decreased rapidly with increase in distance from the store center line so that

the flow over the horizontal tail was relatively unaffected (maximum fluctuations 1 percent of q). It is felt that more fruitful results pertaining to quantitative information on the comparative merits of various store installations may be obtained by surveying the dynamic angularity characteristics of the wake.

CONCLUSIONS

Based on a comparison between high-speed wind-tunnel tests of a model of a fighter-type airplane equipped with external stores and flight tests, it is possible to conclude that:

1. The incremental drag coefficient due to external stores may be used to estimate the maximum Mach number that the airplane may reach in flight when it is equipped with external stores. The data indicate that this estimation is conservative for the five configurations investigated by amounts varying from 0 to 10 percent of the flight-limit Mach number.

2. The free-stream tunnel Mach number corresponding to sonic flow over the lower surface of the wing in the region of the store is a good indication of the lower limit of buffet in flight of the airplane when equipped with external stores.

3. The fluctuations of total pressure over the horizontal tail are not sufficiently large (maximum of 1 percent q_o) to cause buffeting of the airplane.

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REFERENCES

1. Silvers, H. Norman, and Vogler, Raymond D.: Résumé of Wind-Tunnel Data on the Effect of External Stores on Stability of Models of Military Airplanes. NACA RM No. L6K08, 1946.

2. Katzoff, S., and Finn, Robert S.: Effects of External Fuel Tanks and Bombs on Critical Speeds of Aircraft. NACA CB No. L5H27, 1946.

3. Thom, A.: Blockage Corrections in a Closed High-Speed Tunnel. R. & M. No. 2033, British A.R.C., 1943.

TABLE I

Tank	Mounting	Pylon	Extension	Flight-test results		Wind-tunnel results		
				Initial buffet	Limit buffet	$M = 1.0$ (pylon)	M_{cr} (pylon)	M_{c_D}
Lockheed	Left wing	Standard	None	0.51	0.56	0.545	0.640	0.525
Lockheed	Left wing	Standard	6 inch	.60	.64			.575
Lockheed	Left wing	MK 51	None					.60
Lockheed	Left and right wing	Standard	None			.495	.550	.525
Lockheed	Left and right wing	Standard	6 inch					.525
Lockheed	Left and right wing	MK 51	None					(a)
Universal	Left wing	Standard	None	.62	.65	.540	.585	.65
Universal	Left wing	Standard	6 inch			.570	(b)	.675
Universal	Left wing	MK 51	None	None	c.70			.675
Universal	Left and right wing	Standard	None	.54	.58	.535	.610	.575
Universal	Left and right wing	Standard	6 inch					.525
Universal	Left and right wing	MK 51	None					(a)

[a]No clearly defined M_{c_D}.
[b]Insufficient data.
[c]Restricted dive speed of clean airplane 410 knots at 5,000.
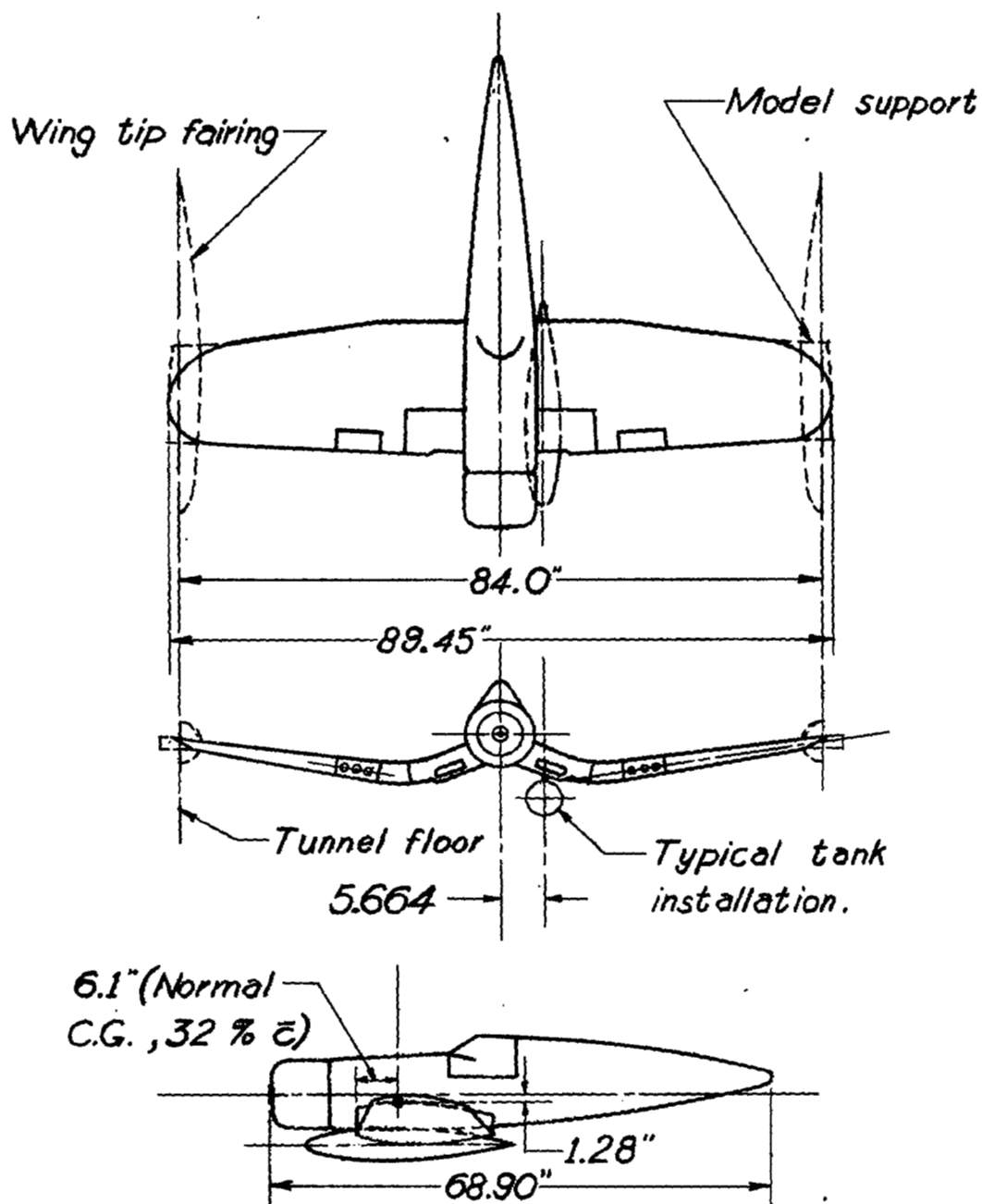
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Figure 1.- Three-view drawing of a model of a fighter-type airplane.
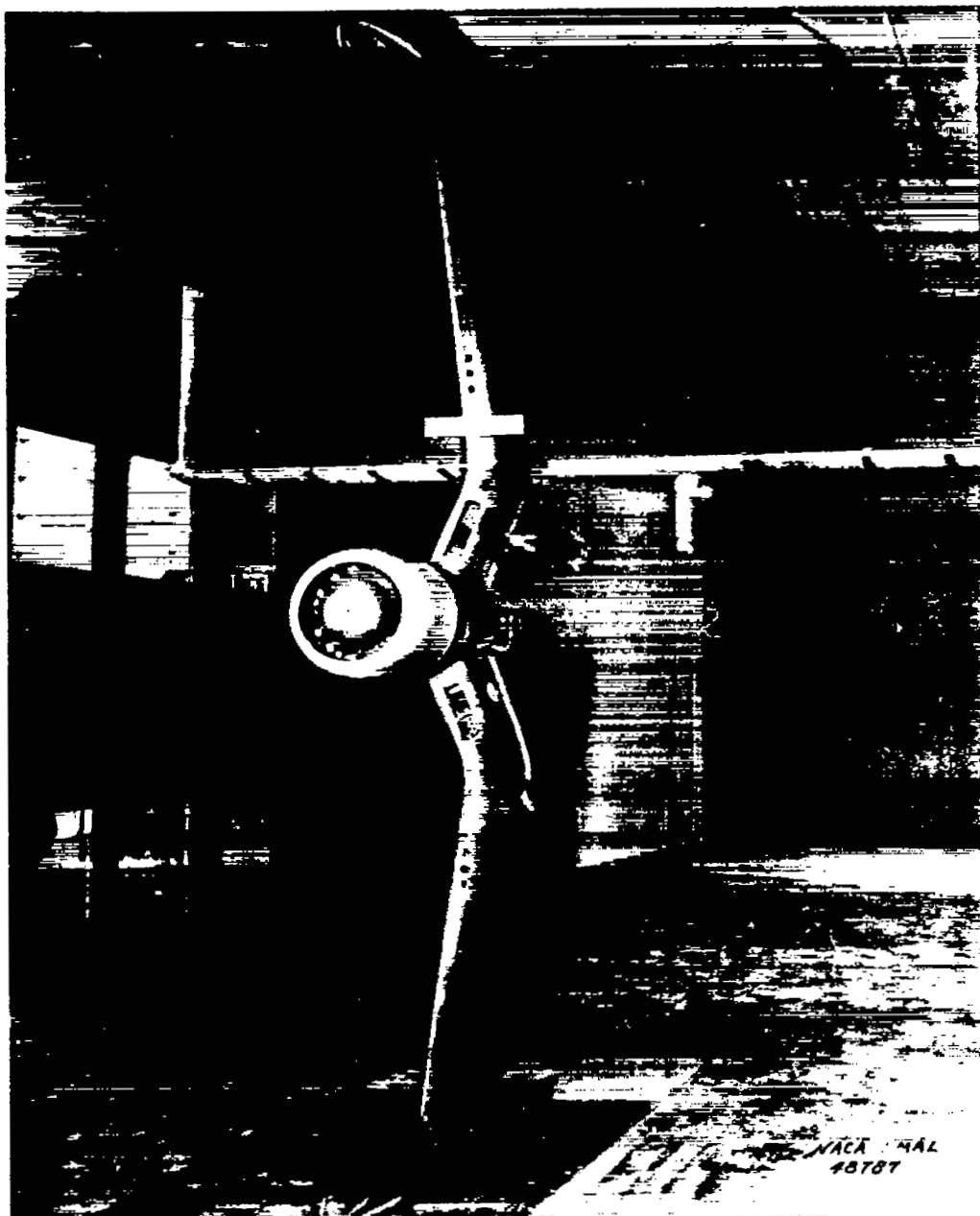
(a) Front view.

Figure 2.- Photograph of a model of a fighter-type airplane mounted
    in the high-speed 7- by 10-foot tunnel showing external store
    installation.

(b) Rear view.

Figure 2.- Concluded.

Pylon ℄ Sta. 5.664

Sta. 8

Sta. 5 1/4

Sta. 9

Orifices Location	
Number	Percent c̄
A	11.3
B	18.2
C	25.7
D	53.0
E	69.5
F	85.5
G	4.6
I	11.3
K	5.8
L	13.0
N	33.9
O	44.3
P	59.4
R	85.2
H	7.7

Orifices on bottom surface of wing

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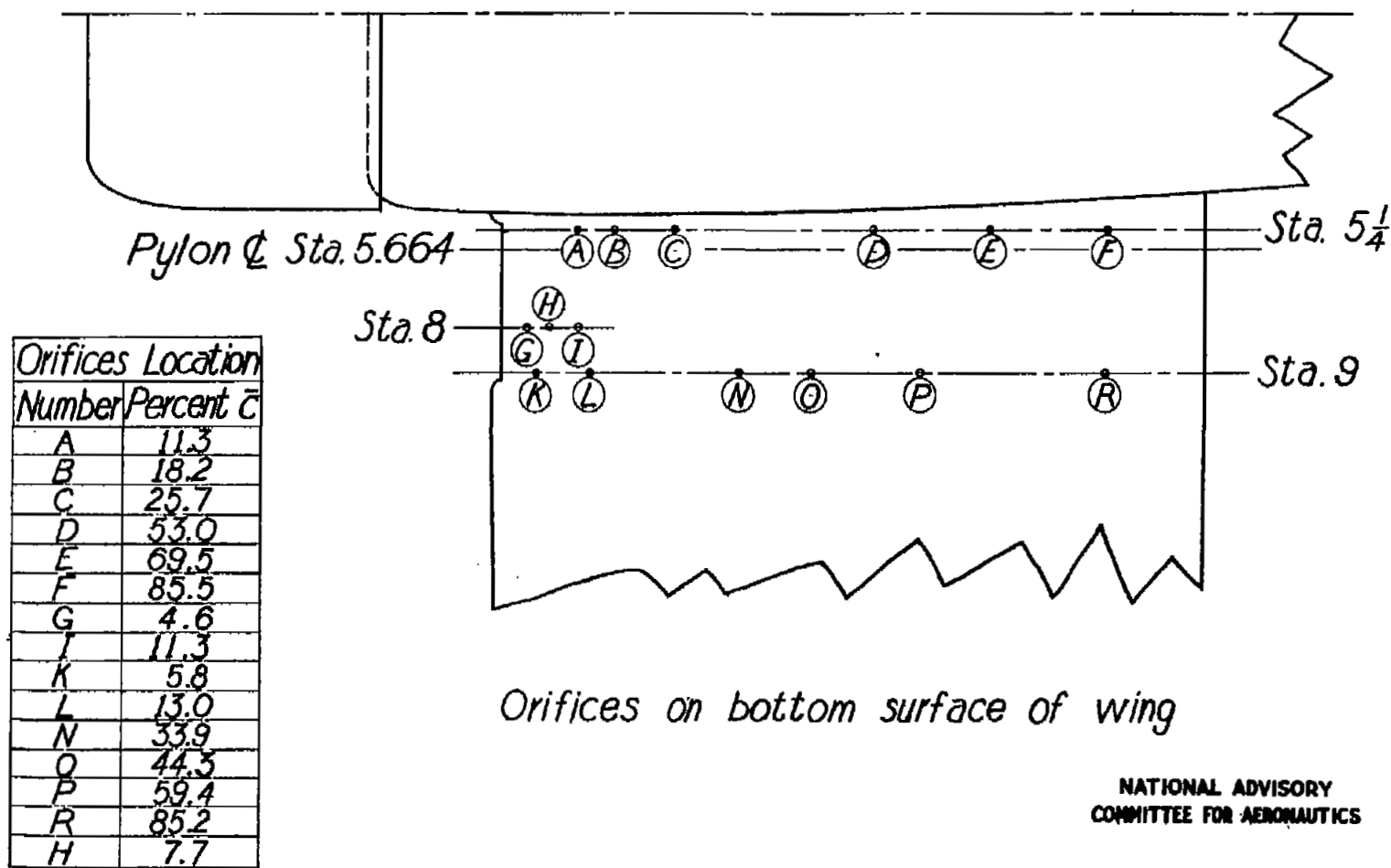
Figure 3.- Location of wing pressure orifices on a model of a fighter-type airplane.

pressure orifice

45°

typical section

wing leading edge

-.15
42
0.7
20.8
37.4
48.4
Standard pylon-tank juncture
66.0
76.2

Lockheed Tank

wing leading edge

-4.5
1.7
8.0
20.4
29.2
39.1
51.5
Standard pylon-tank juncture
70.9
86.0

Universal Tank

Ordinates for Lockheed tank	
x	y
.0	0
1.25	±2.46
2.5	±3.40
5.0	±4.47
7.5	±5.99
10	±6.94
15	±8.38
20	±9.47
30	±10.92
40	±11.52
50	±11.46
60	±10.41
70	±8.48
80	±5.85
90	±3.31
95	±2.12
100	±1.08
L.E. radius 3.03	

Ordinates for Universal tank	
x	y
.0	0
1.25	±2.02
2.5	±2.98
5.0	±4.25
7.5	±5.15
10	±5.87
15	±6.94
20	±7.75
30	±8.74
40	±9.16
50	±9.16
60	±8.70
70	±7.64
80	±5.93
90	±4.09
95	±3.15
100	0
L.E. radius 2.25	
T.E. radius 2.72	

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Figure 4.- Location of pressure orifices on models of Lockheed and
Universal auxiliary fuel tanks as tested on a model of a fighter-type
airplane.

pressure orifice locations in percent wing chord
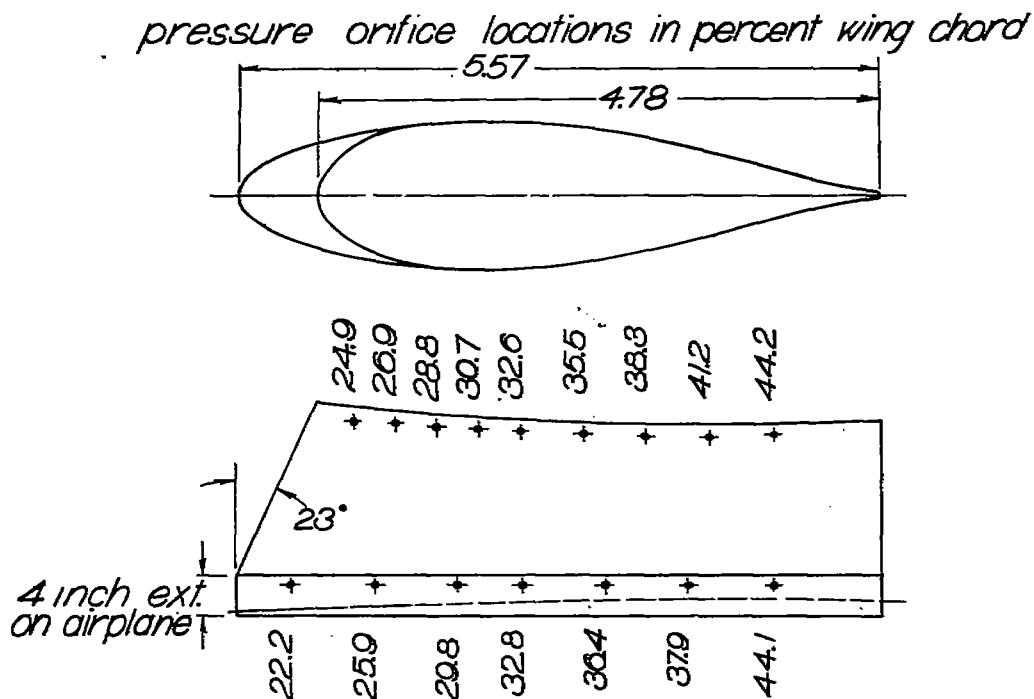
Pylon Ordinates		
X	upper juncture	lower juncture
0	0	0
1.25	±2.69	±3.77
2.5	±4.13	±5.65
5.0	±5.93	±7.95
7.5	±7.00	±8.78
10	±8.27	±11.30
15	±9.70	±12.98
20	±10.78	±13.80
30	±11.86	±14.01
40	±12.02	±13.60
50	±11.49	±12.33
60	±10.23	±10.25
70	±7.90	±7.95
80	±5.21	±5.02
90	±2.33	±2.30
95	±1.26	±1.05
100	±0.36	±0.42
L.E. radius	3.95	5.65

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Figure 5.- Location of pressure orifices on a model of a Standard
pylon as tested on a model of a fighter-type airplane.
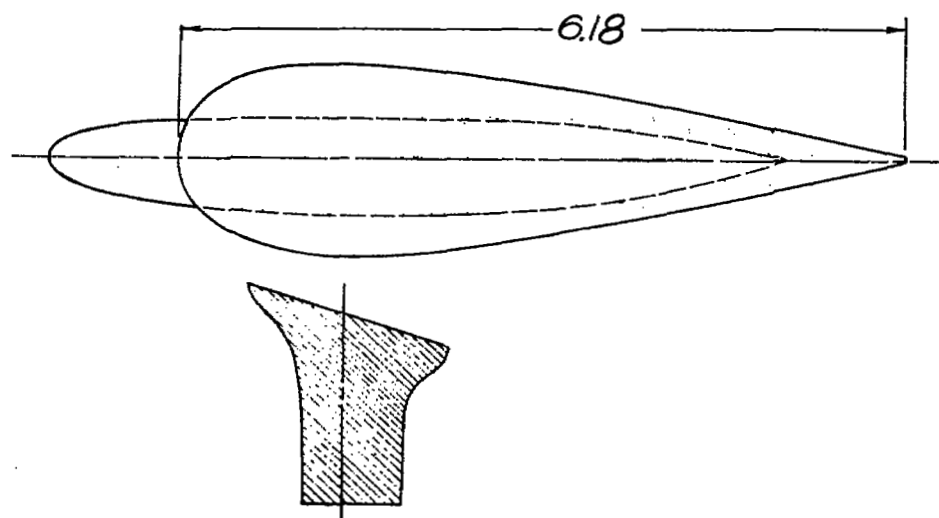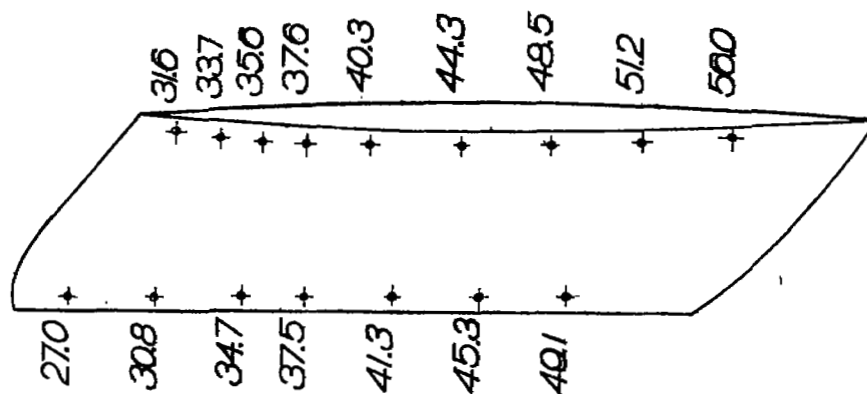
6.18

typical section

316 337 350 37.6 40.3 44.3 48.5 512 560

270 308 34.7 37.5 41.3 45.3 49.1

pressure orifice locations in percent wing chord

Figure 6.-   Location of pressure orifices on a model of a Mark 51
pylon as tested on a model of a fighter-type airplane.
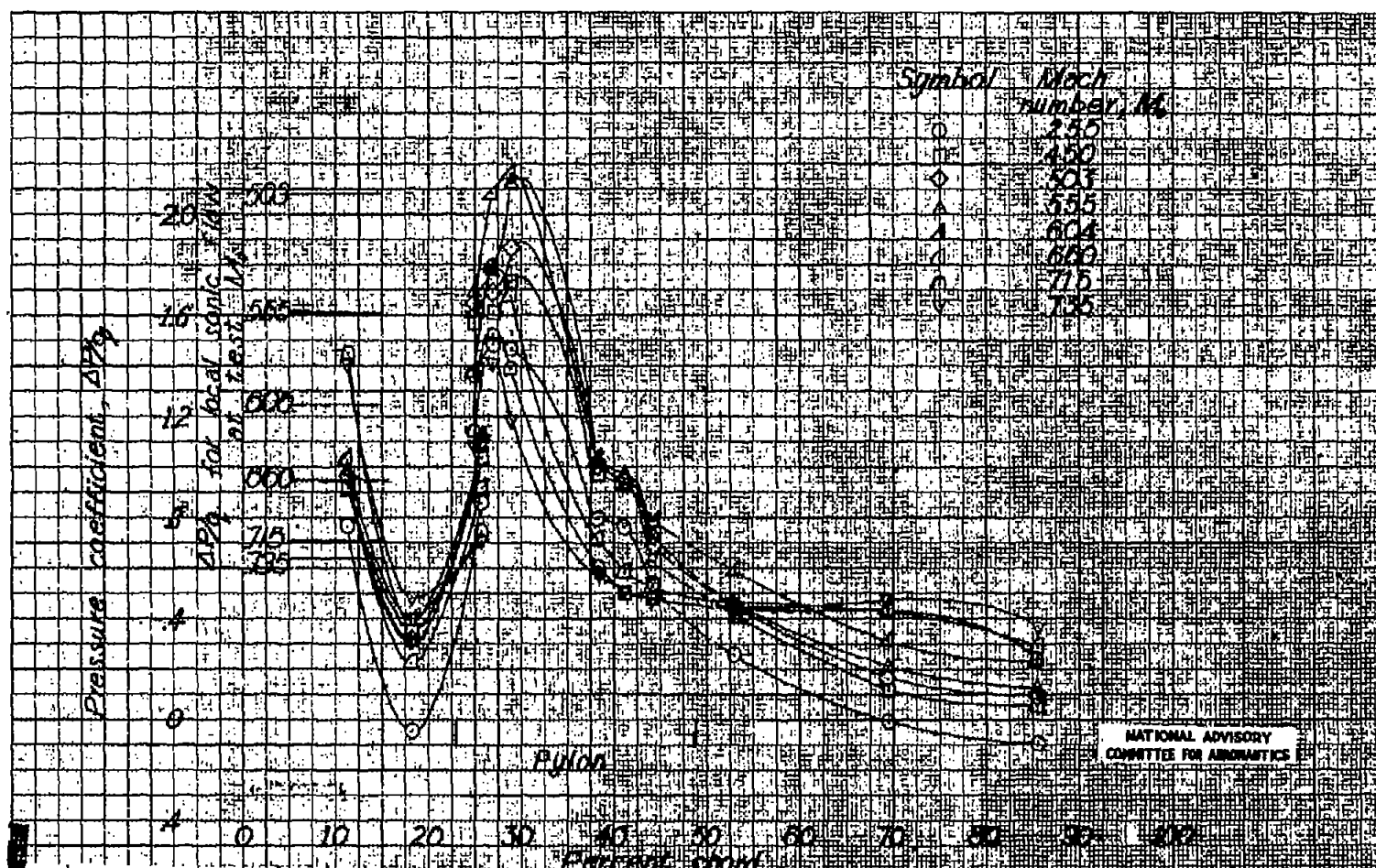
(a) Lockheed tank on left standard pylon.

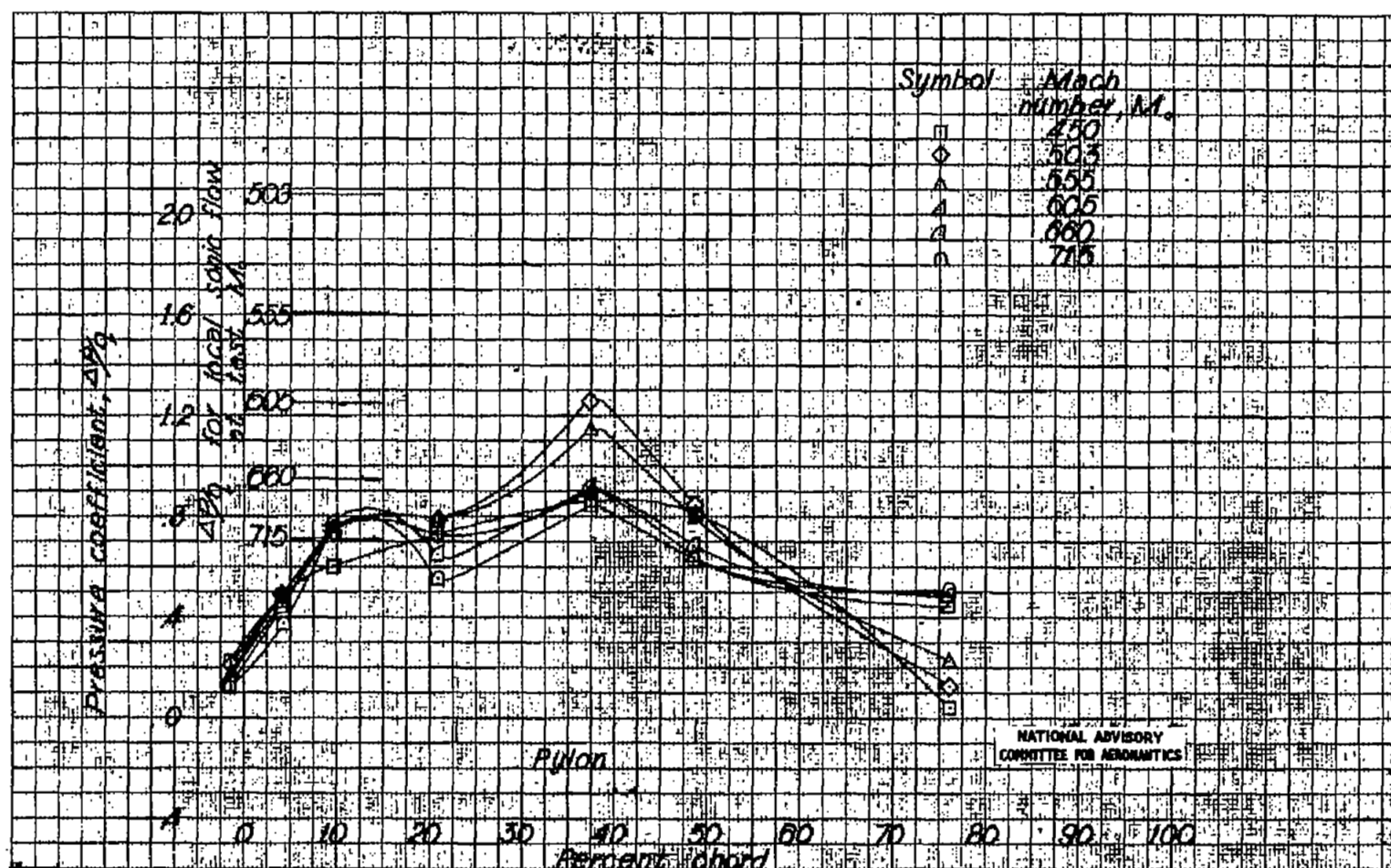Figure 7.- Pressure distribution on the lower surface of the wing of a model of a fighter-type airplane with external stores. Inboard orifice row, $\alpha = -2.75^\circ$.
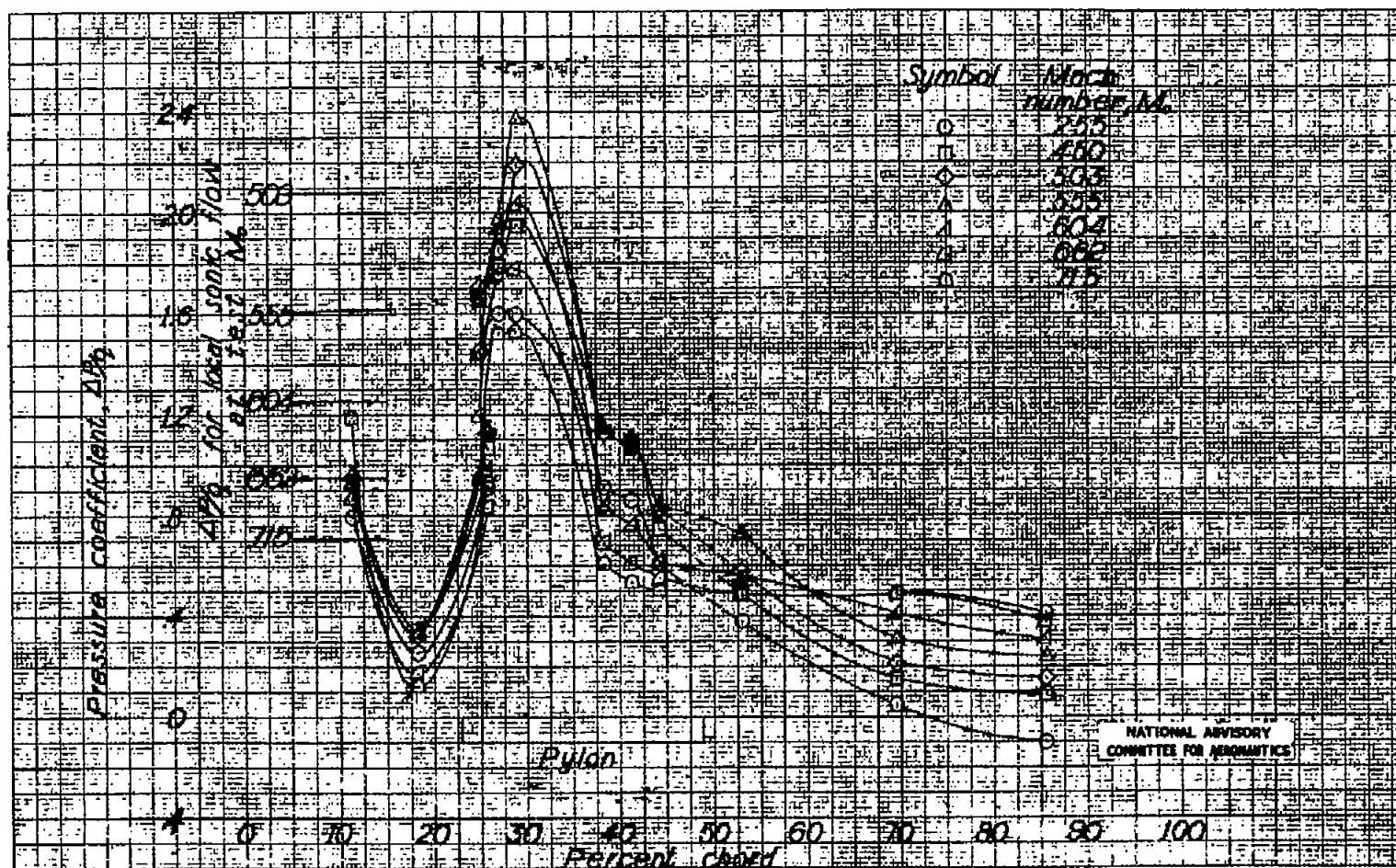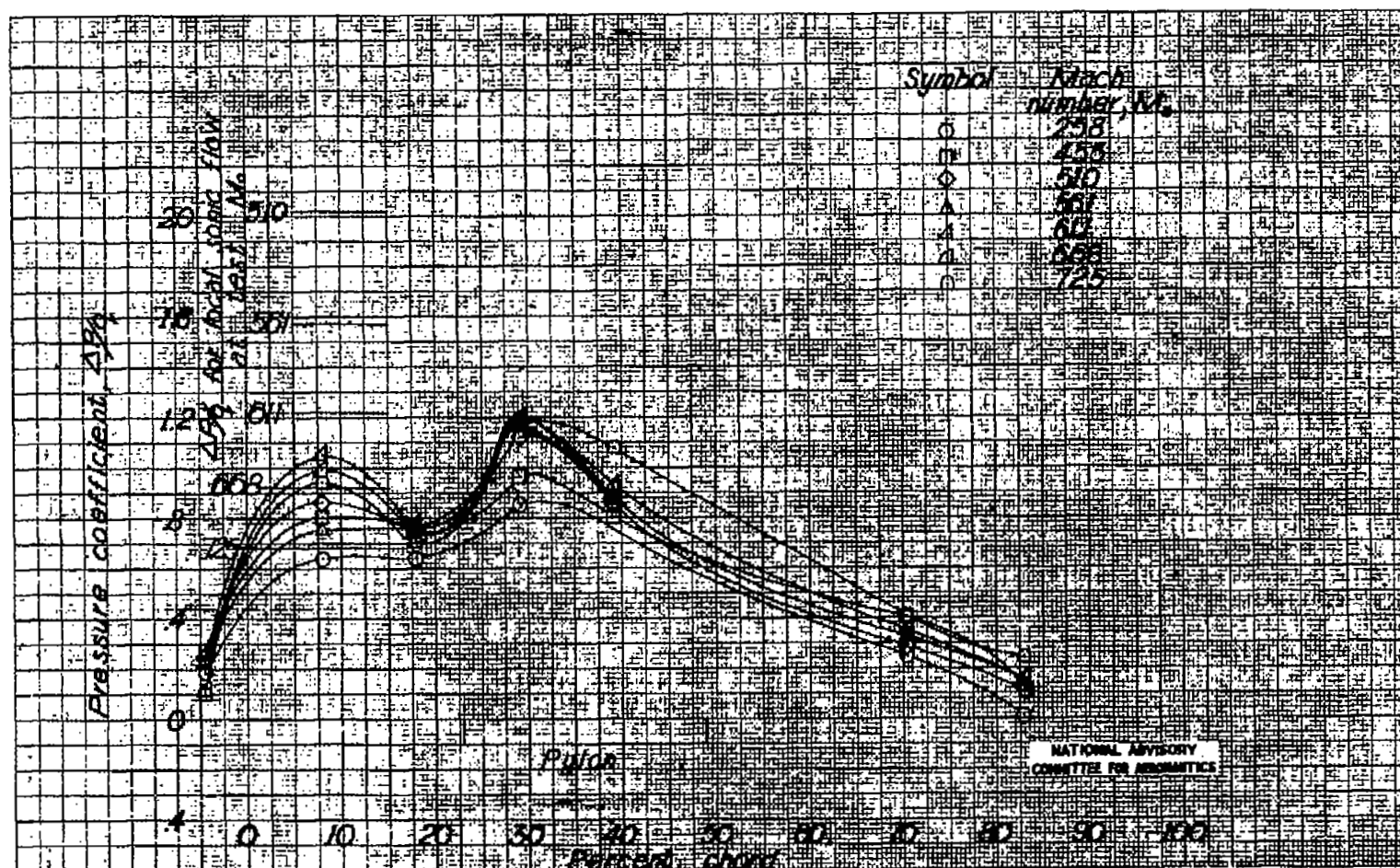
(b)   Lockheed tanks on left and right standard pylon.

Figure 7.-   Continued.
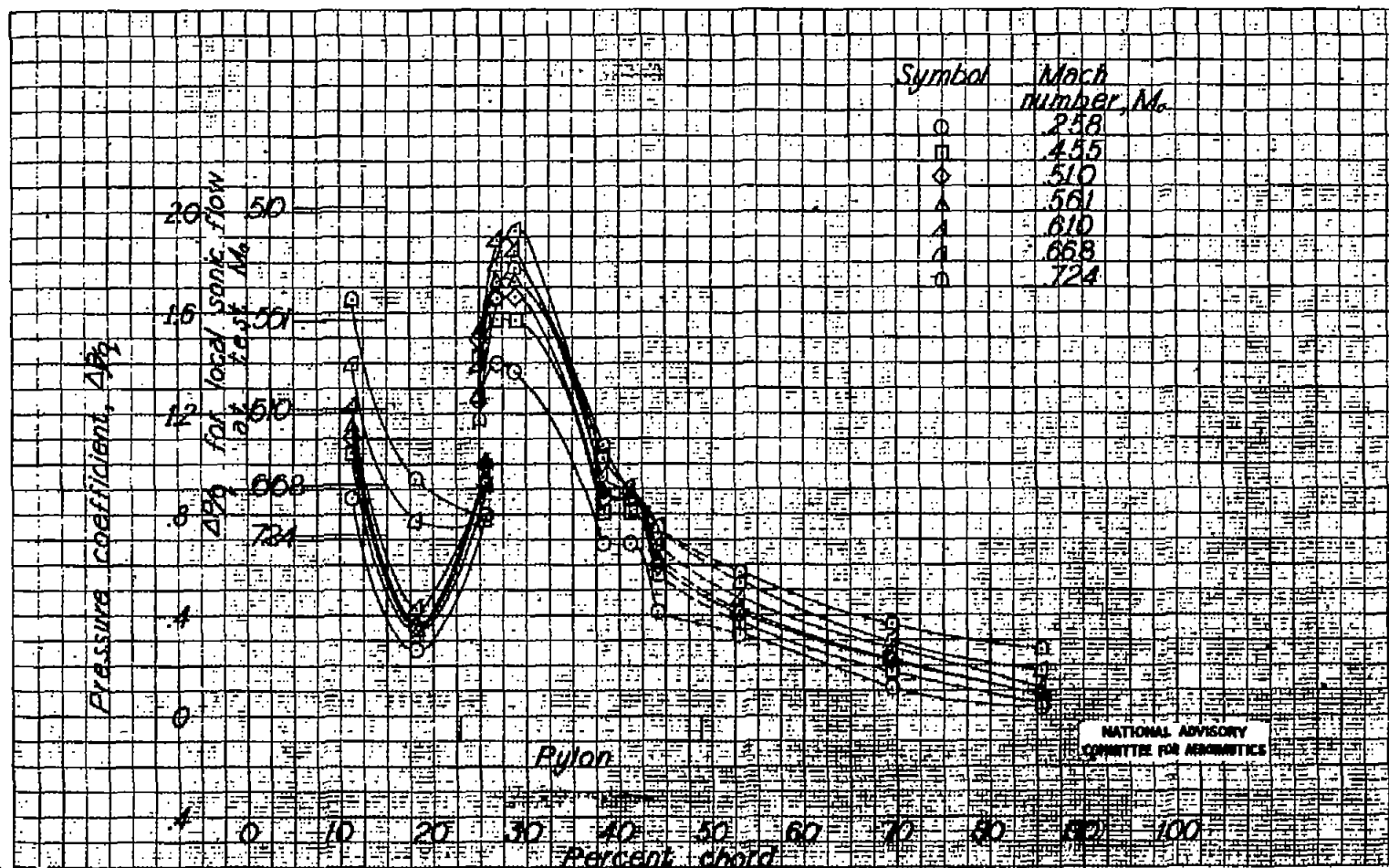
(c) Universal tank on left standard pylon.
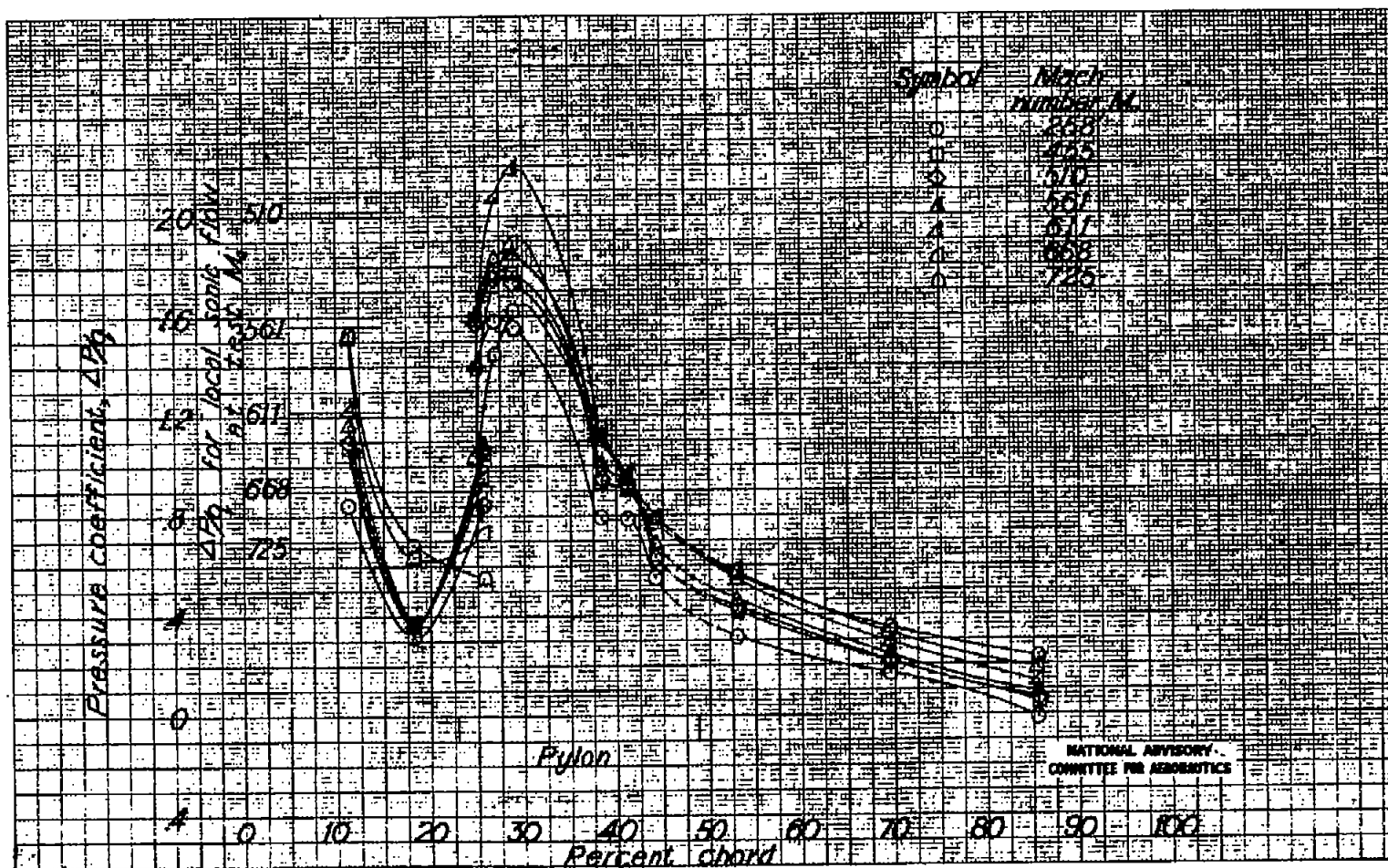
Figure 7.- Continued.

(d) Universal tanks on left and right standard pylons.

Figure 7.- Continued.
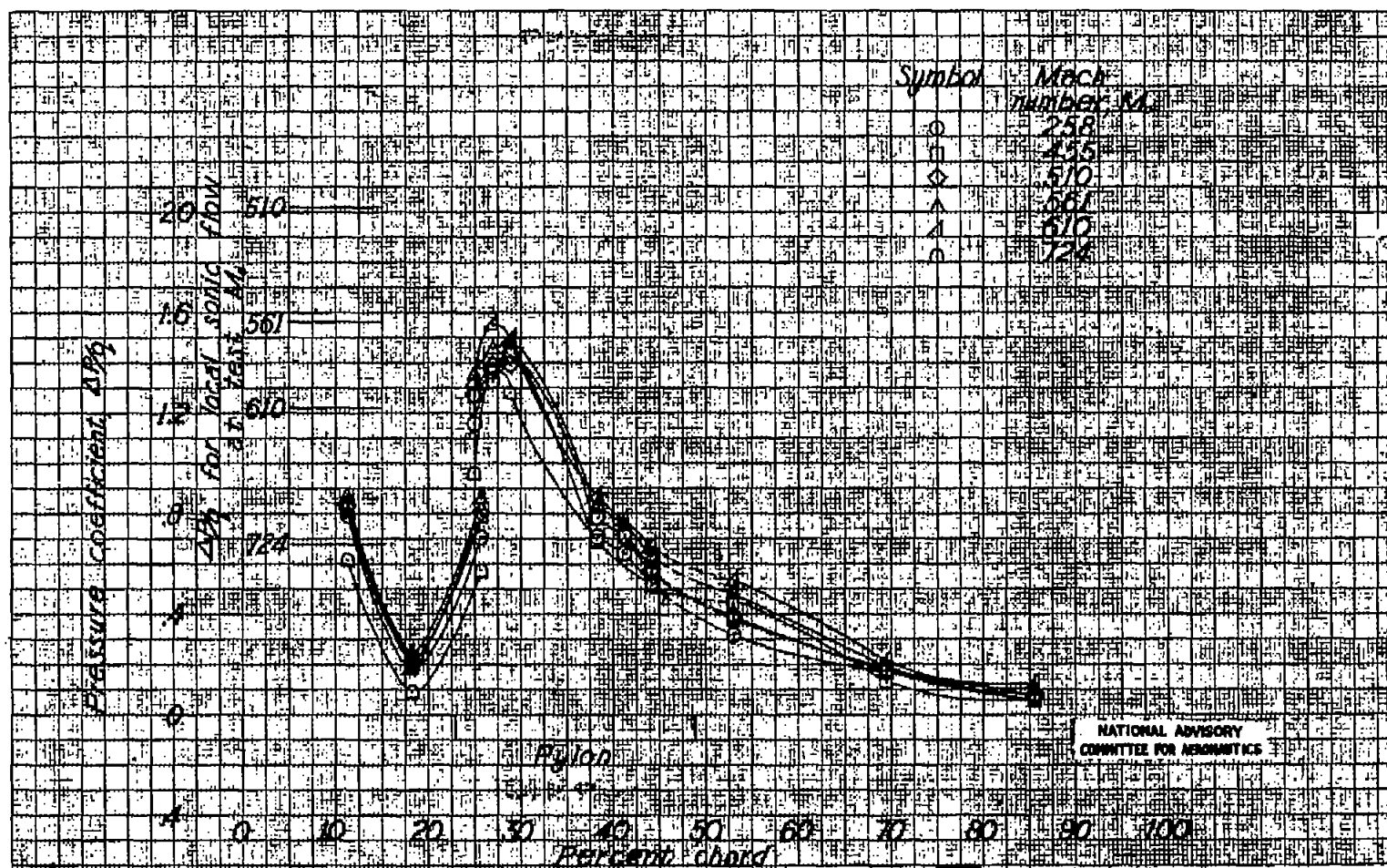
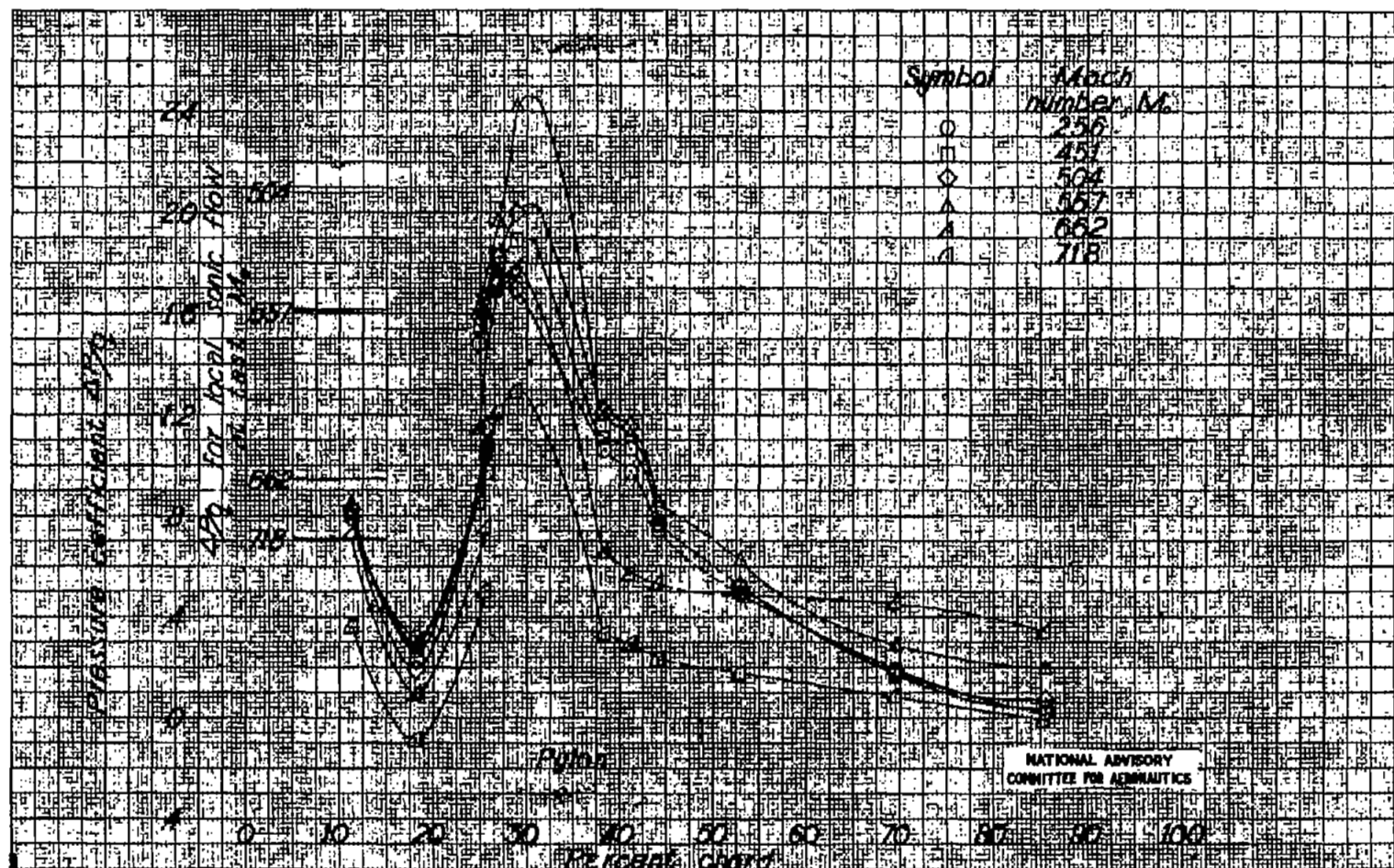(e) Universal tank on left standard pylon with 6-inch extension.

Figure 7.- Concluded.
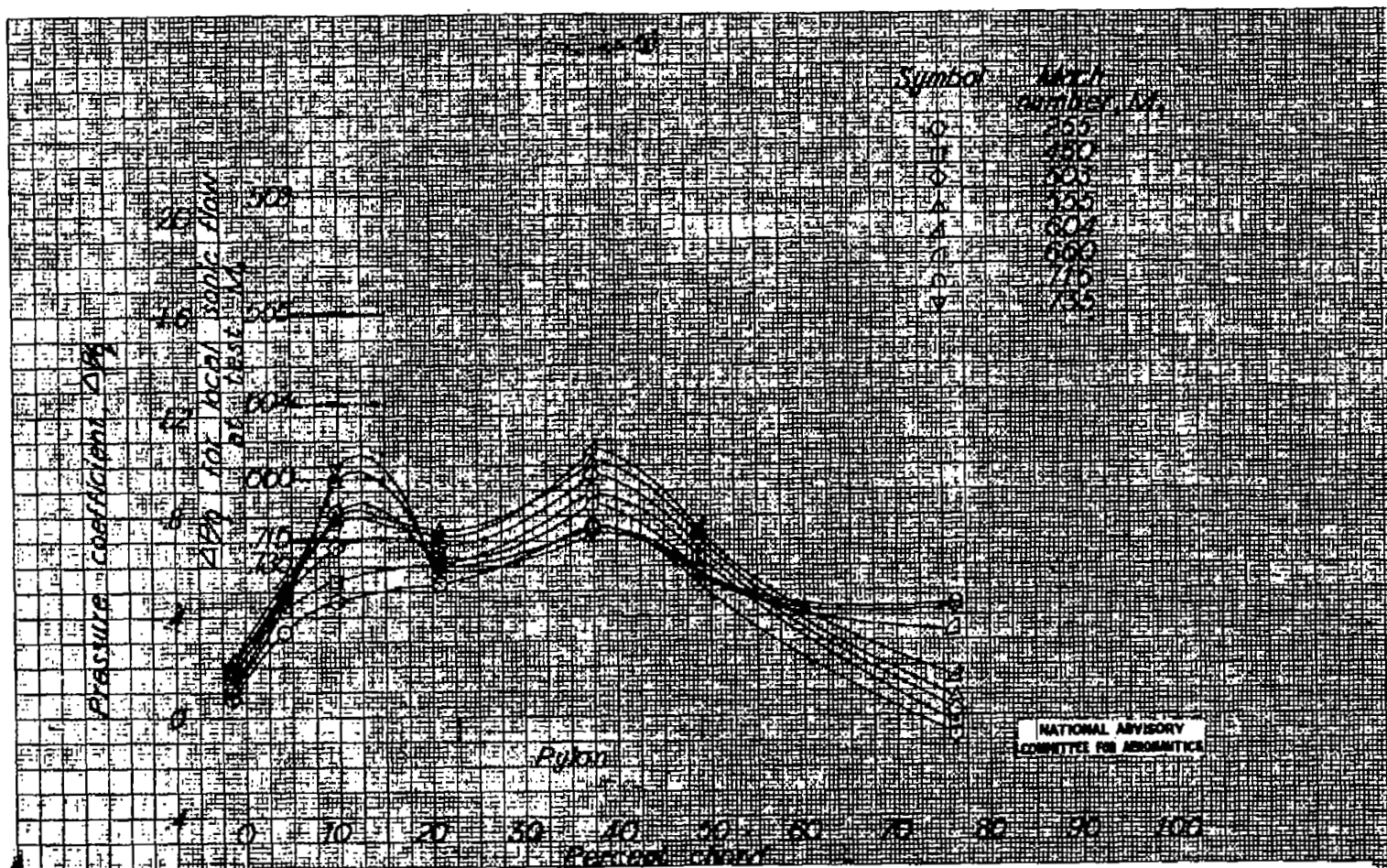
(a) Lockheed tank on left standard pylon.

Figure 8.- Pressure distribution over droppable fuel tanks mounted on the wing of a model of a fighter-type airplane. $\alpha = -2.75^{\circ}$.
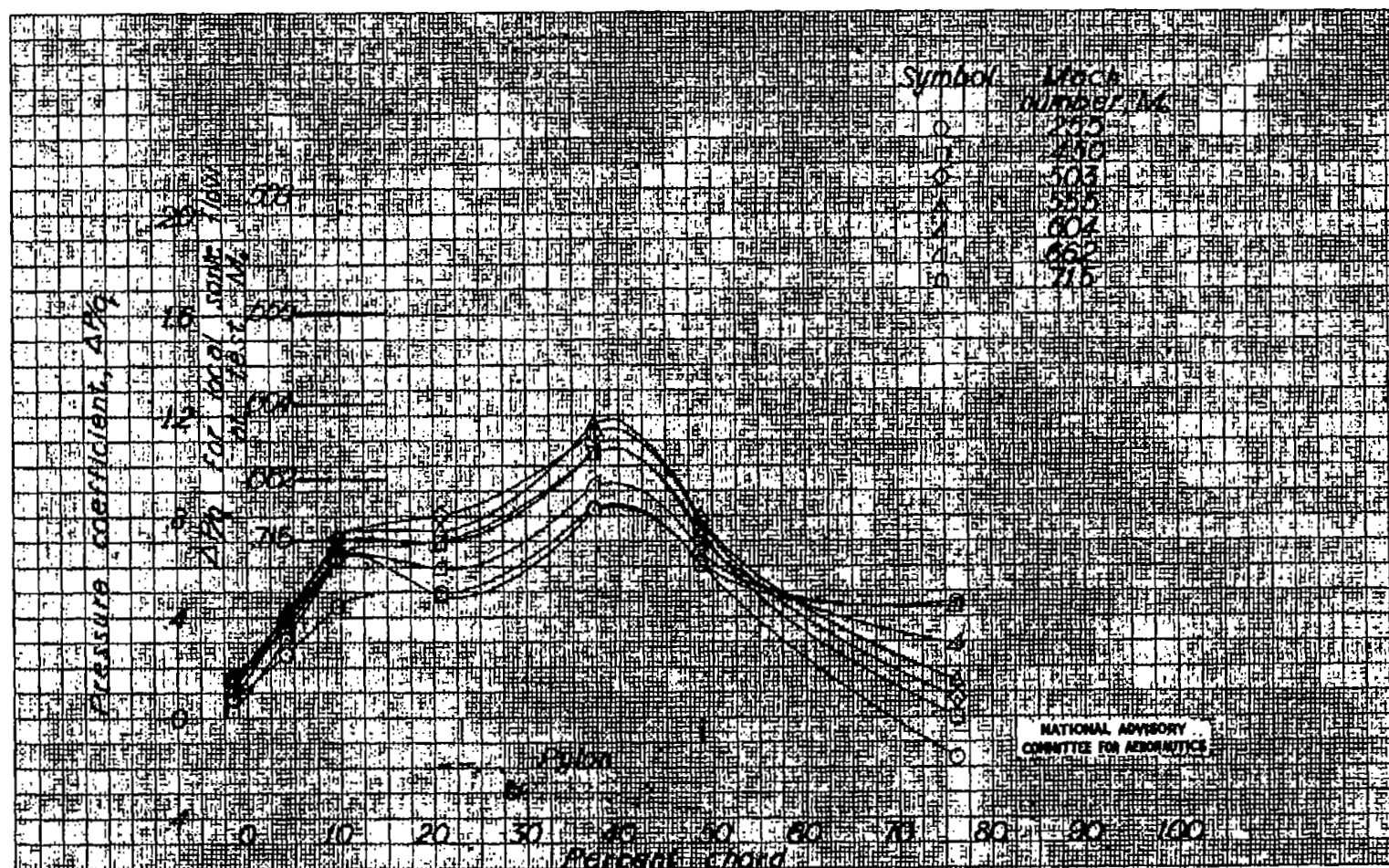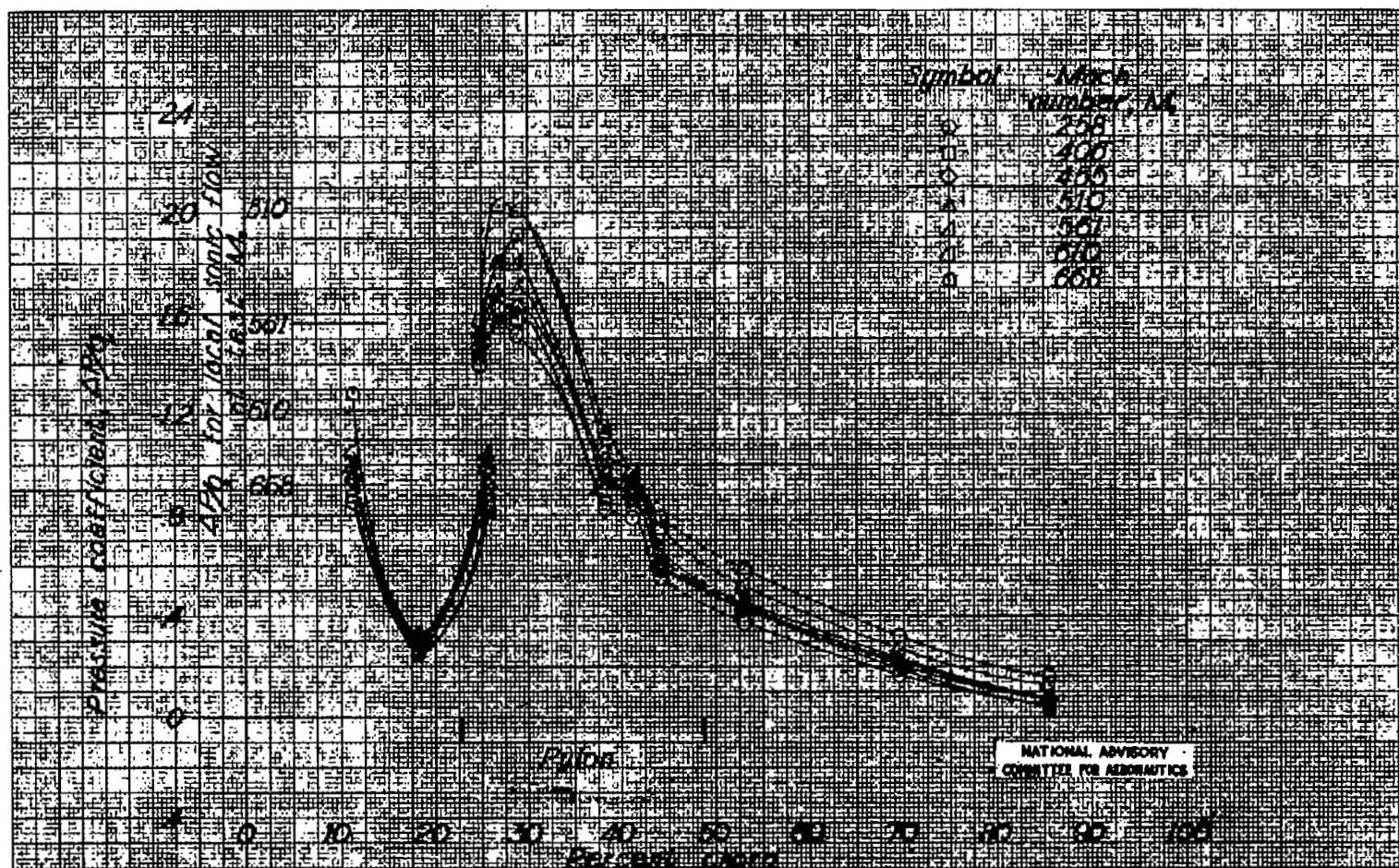
(b) Lockheed tanks on left and right standard pylons.
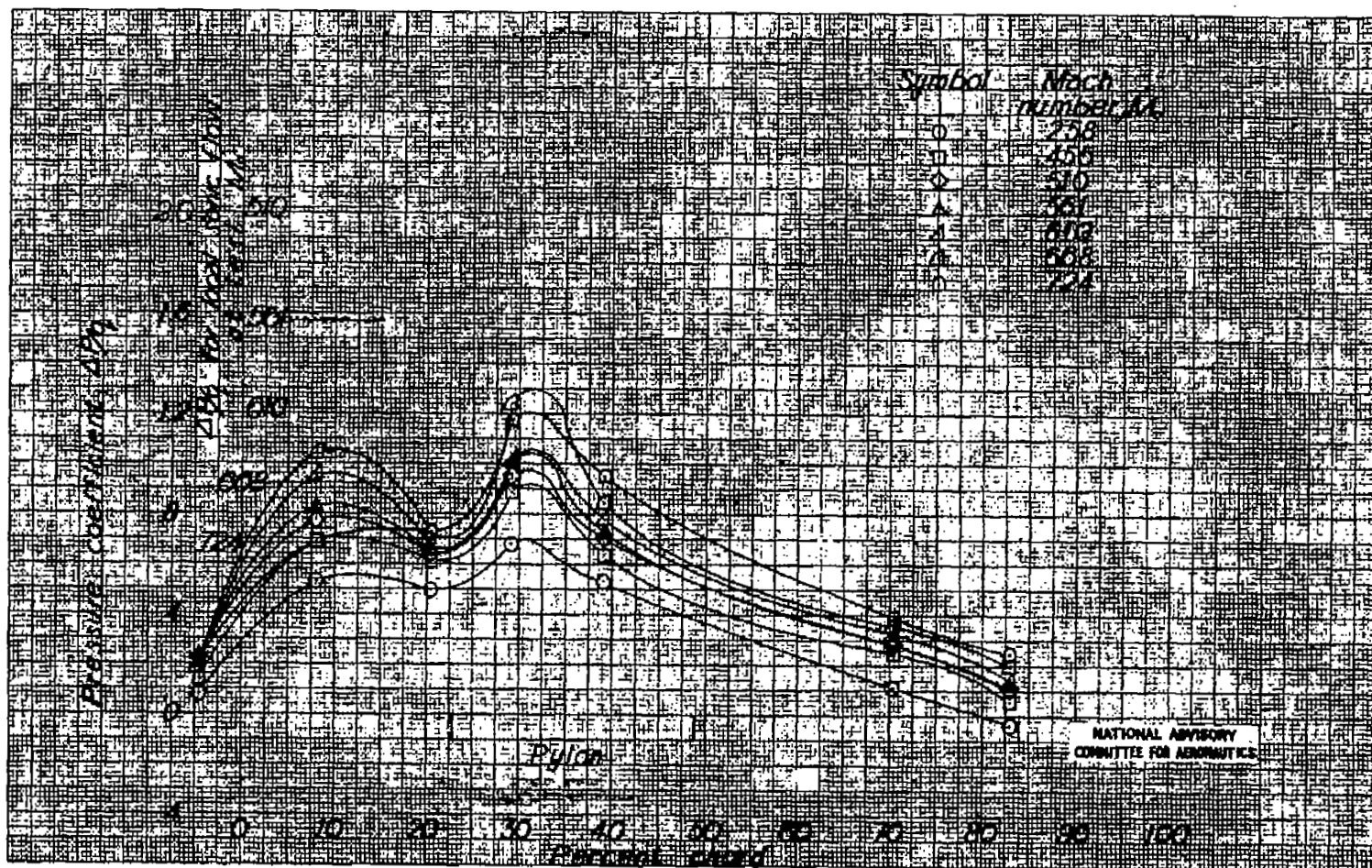
Figure 8.- Continued.

(c) Universal tank on left standard pylon.

Figure 8.- Continued.
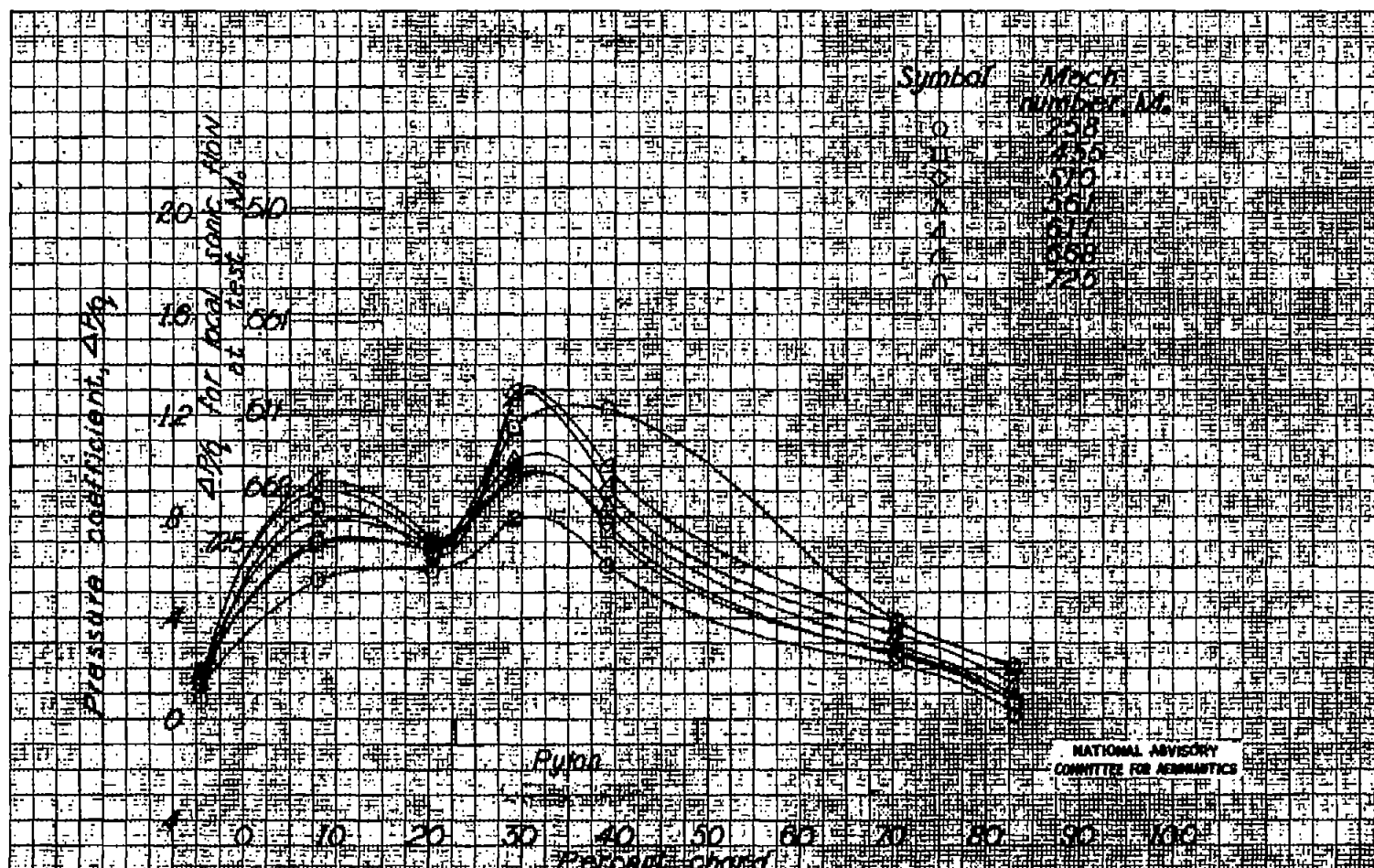
(d) Universal tanks on left and right standard pylons.

Figure 8.- Continued.
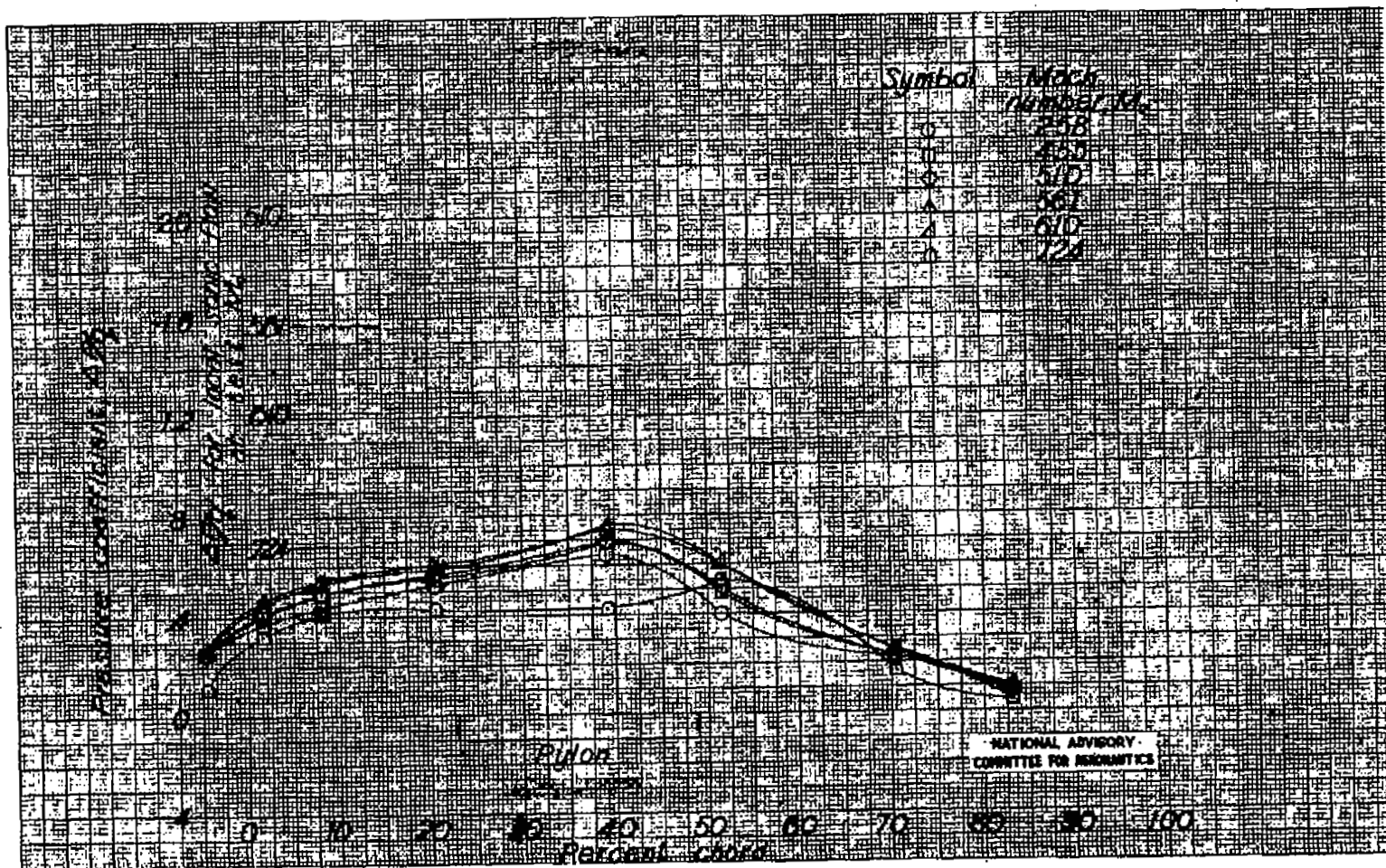
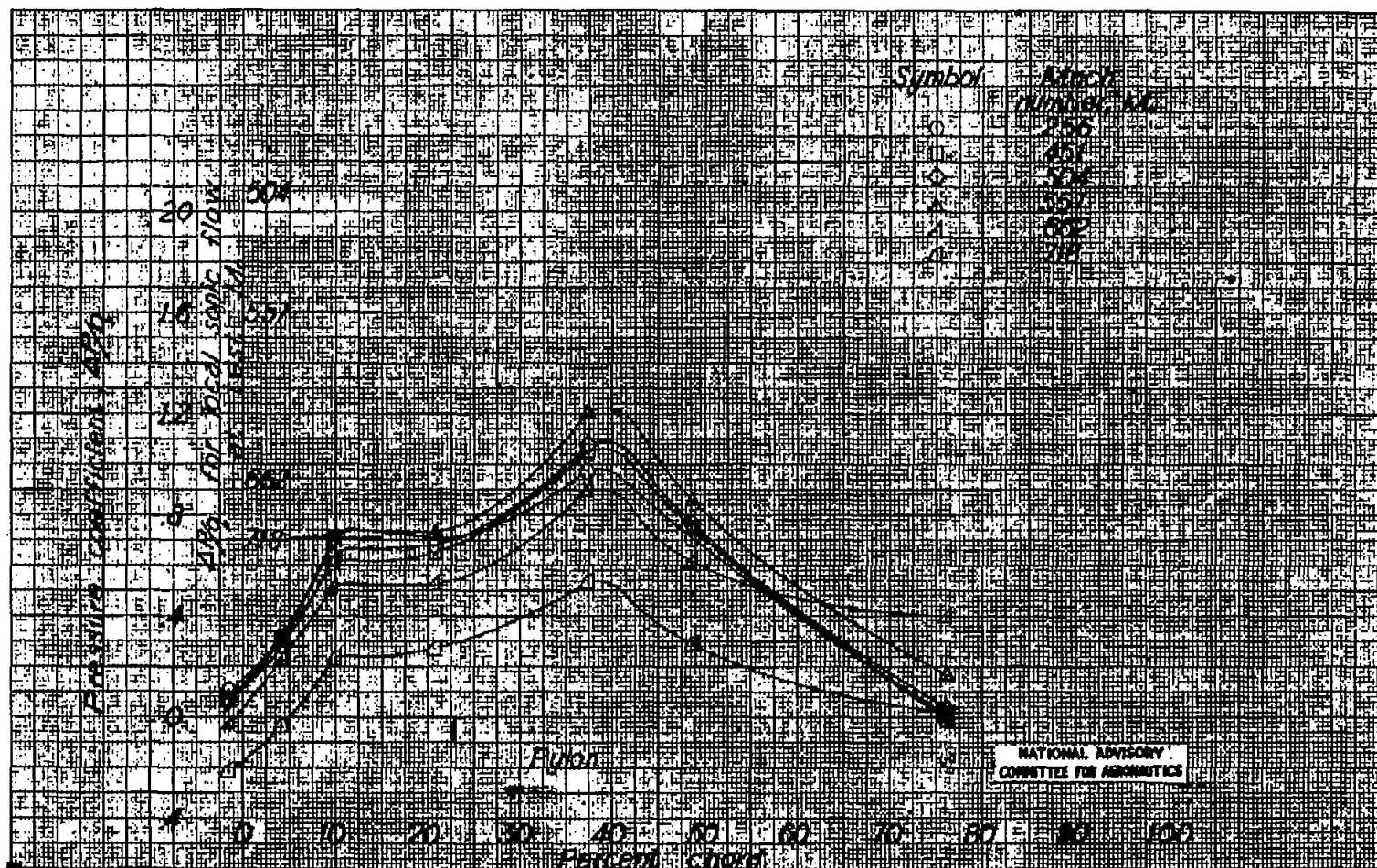(e) Universal tank on left standard pylon with 6-inch extension.

Figure 8.- Concluded.
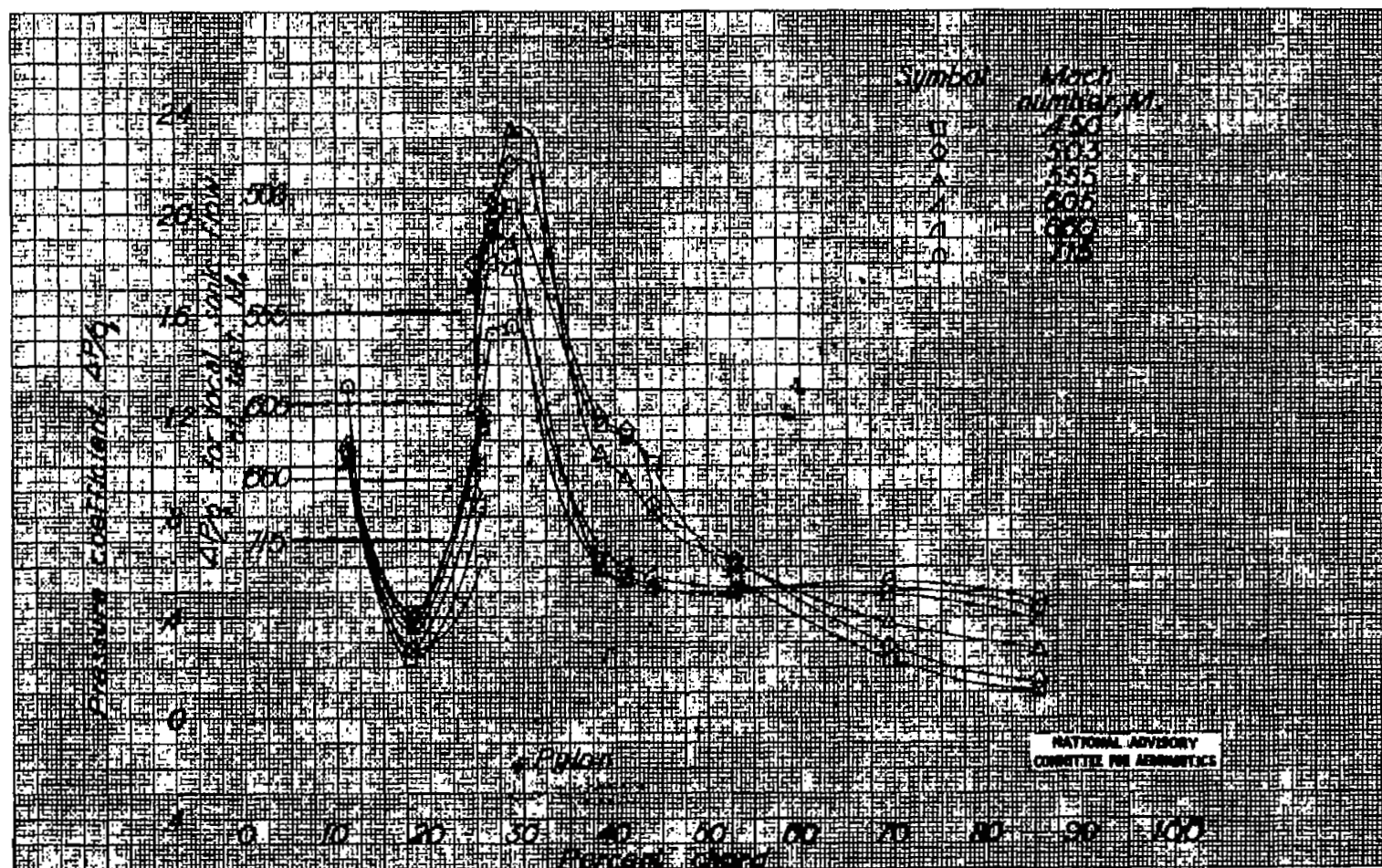
(a) Lockheed tanks on left and right standard pylons.

Figure 9.- Pressure distribution on the lower surface of the wing of a model of a fighter-type airplane with external stores. Inboard orifice row, $\alpha = -3.5°$.
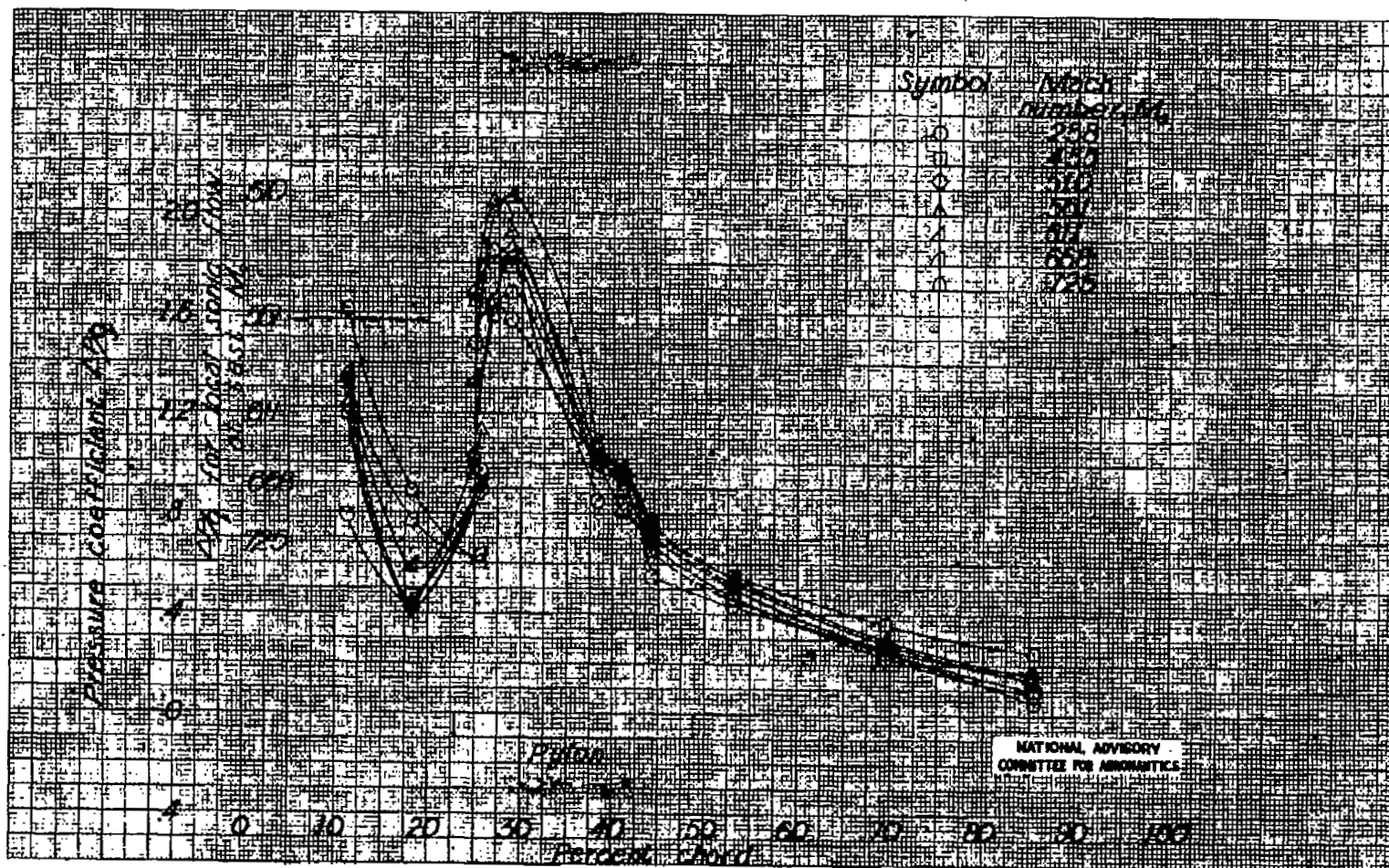
(b) Universal tanks on left and right standard pylons.

Figure 9.- Concluded.

(a) Lockheed tanks on left and right standard pylons.

Figure 10.- Pressure distribution over droppable fuel tanks mounted on the wing of a fighter-type airplane. $\alpha = -3.5^{\circ}$.
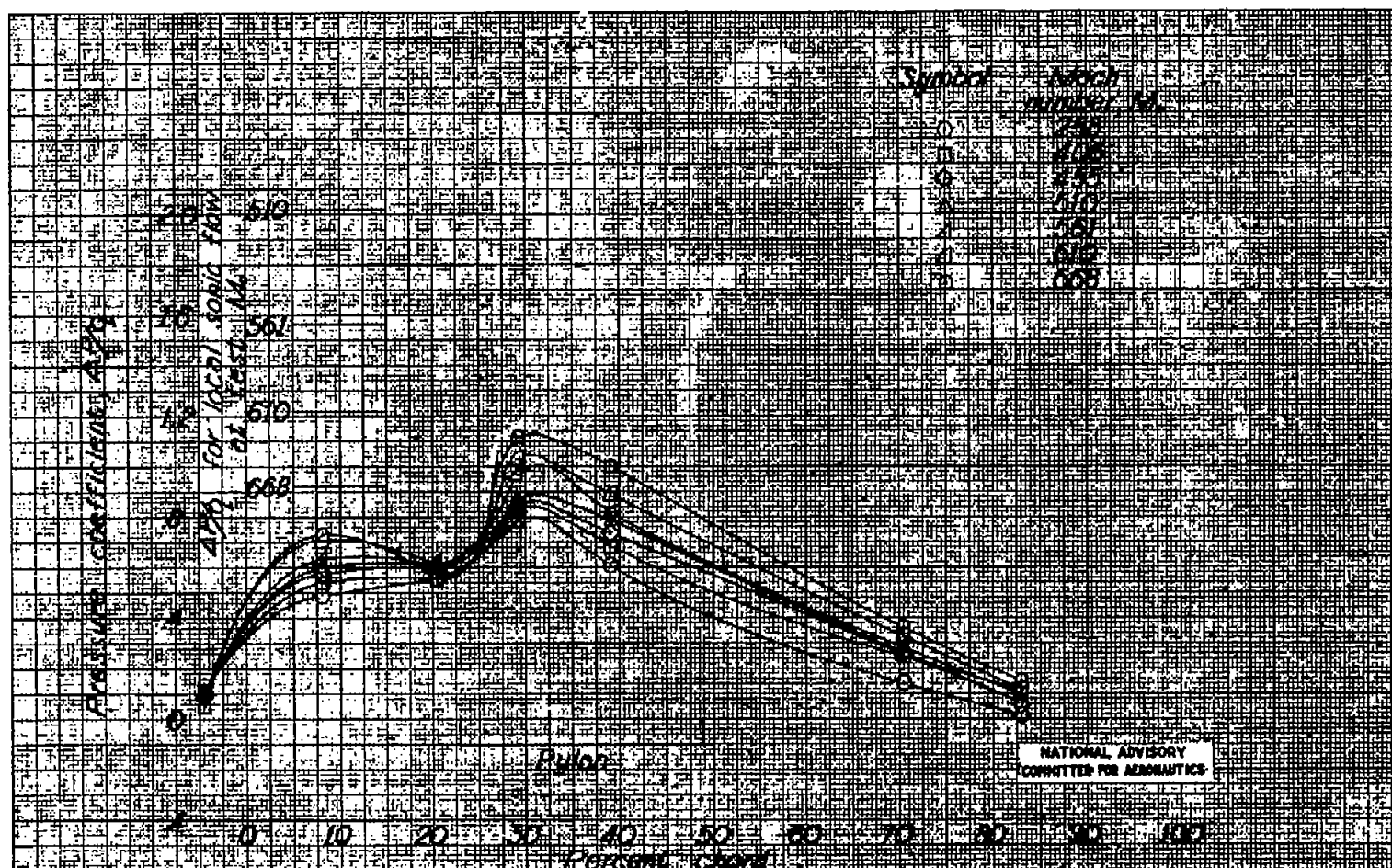
(b) Universal tanks on left and right standard pylons.

Figure 10.- Concluded.

(a) Lockheed tanks on left and right standard pylons.

Figure 11.- Pressure distribution on the lower surface of the wing of a model of a fighter-type airplane with external stores. Inboard orifice row, $\alpha = -1.5^{\circ}$.

(b) Universal tanks on left and right standard pylons.
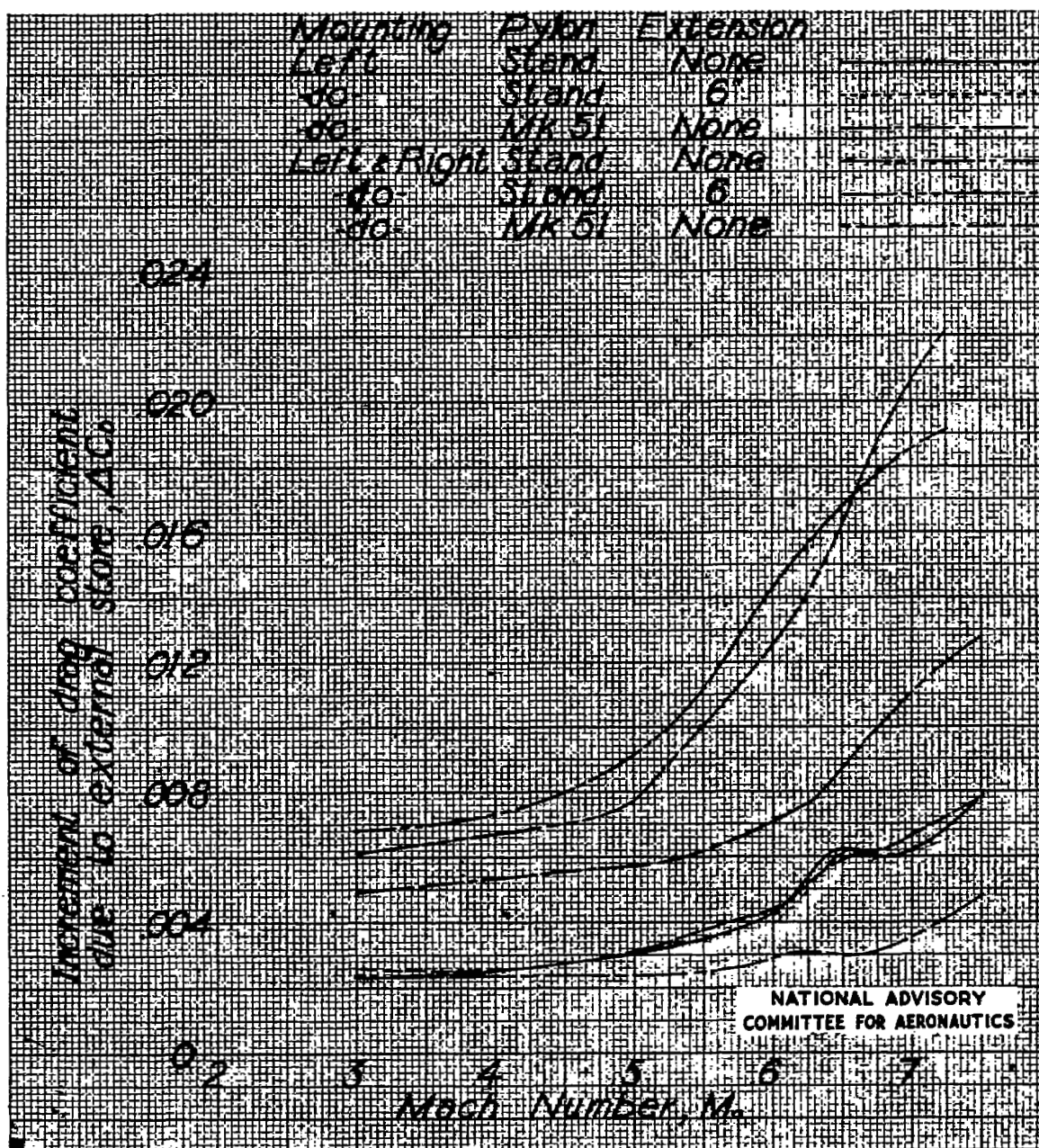
Figure 11.- Concluded.

(a) Lockheed tanks on left and right standard pylons.

Figure 12.- Pressure distribution over droppable fuel tanks mounted on the wing of a model of a
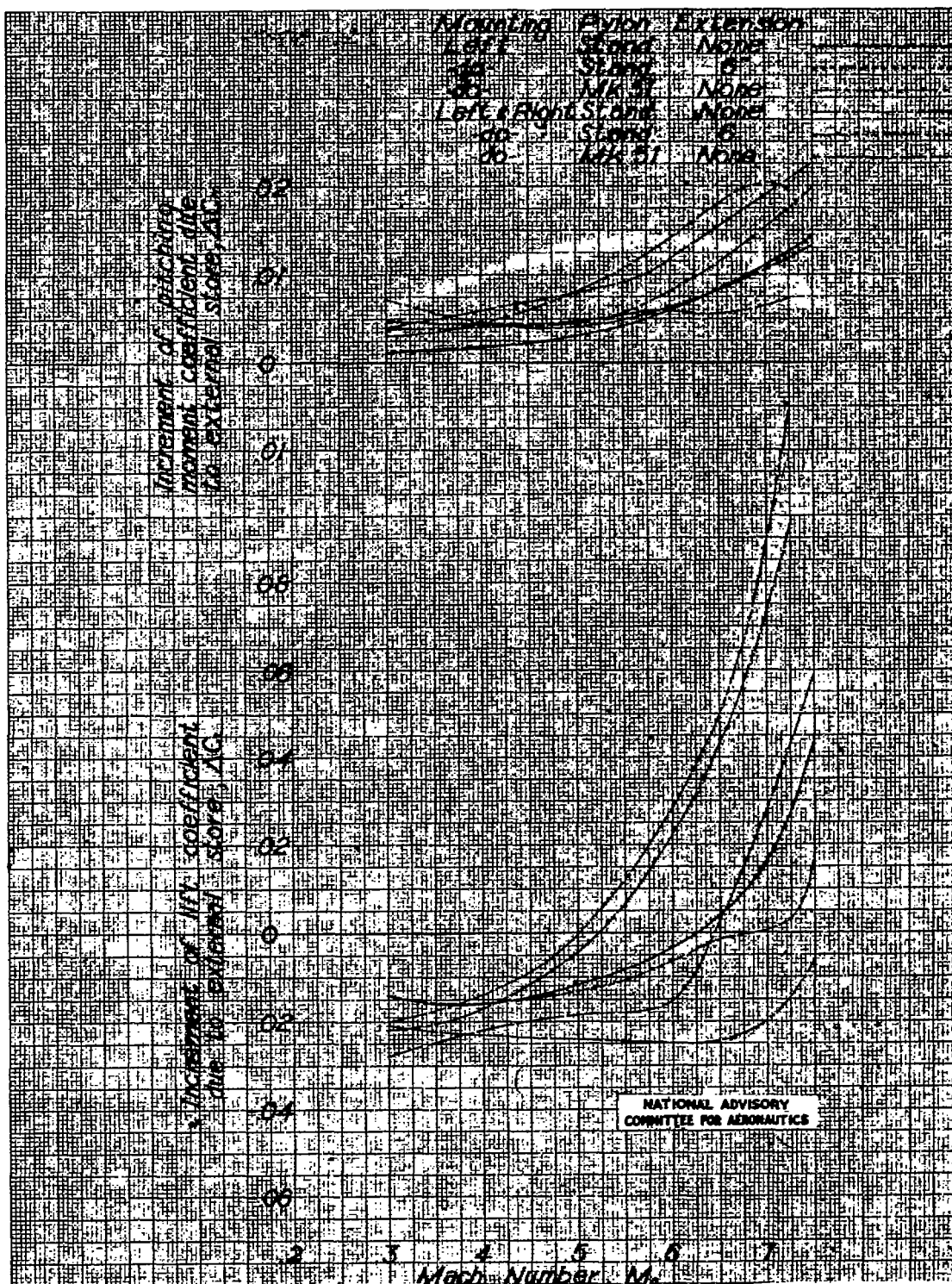fighter-type airplane, α = -1.5°.

(b) Universal tanks on left and right standard pylons.

Figure 12.- Concluded.

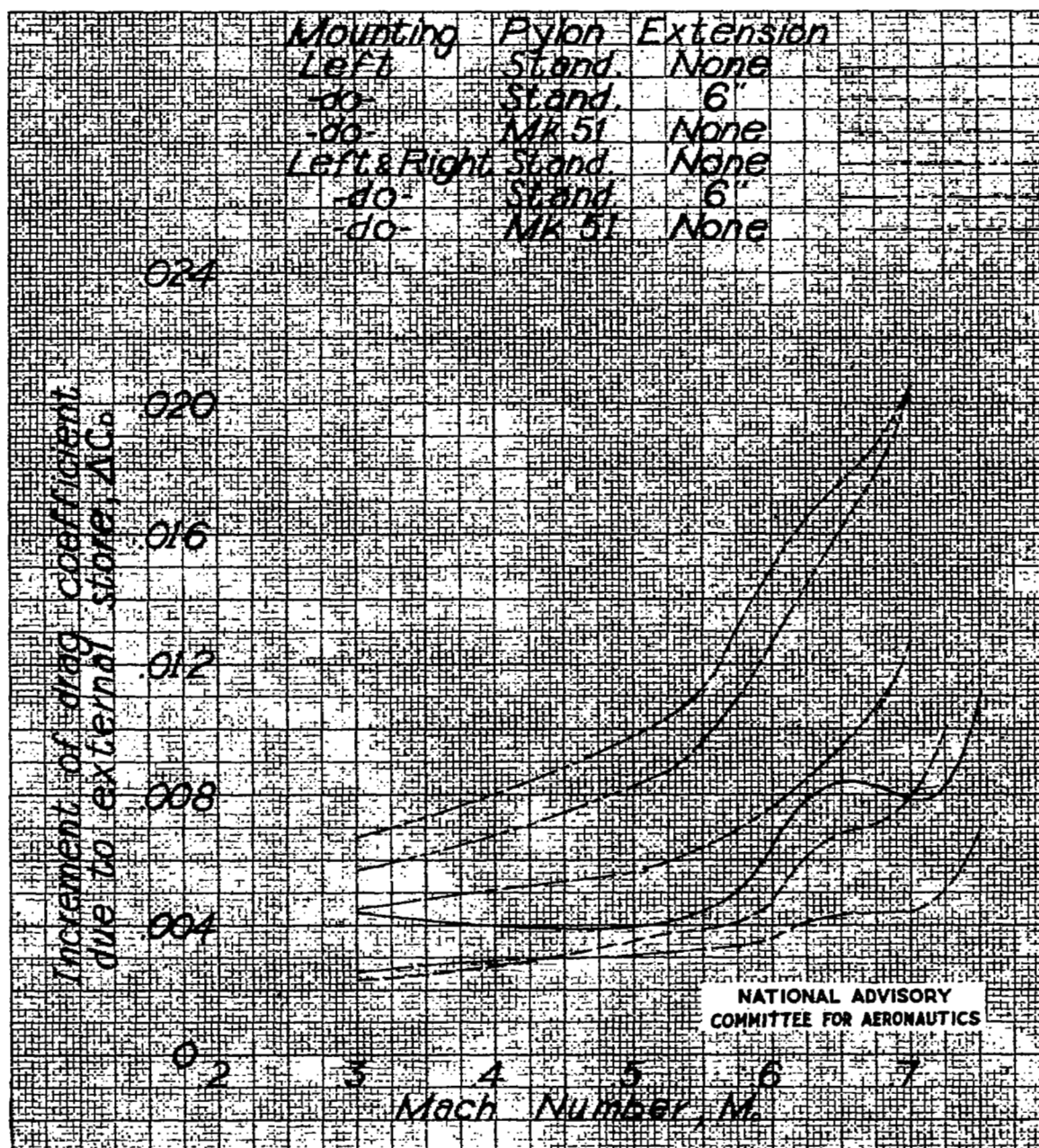| Mounting	Pylon	Extension
Left	Stand	None
do	Stand	6"
do	Mk 51	None
Left & Right	Stand	None
do	Stand	6"
do	Mk 51	None

(a) $\alpha = -3.5^{O}$.

Figure 13.- Effect of various pylon configurations on the aerodynamic
characteristics of a model of a fighter-type airplane with Lockheed
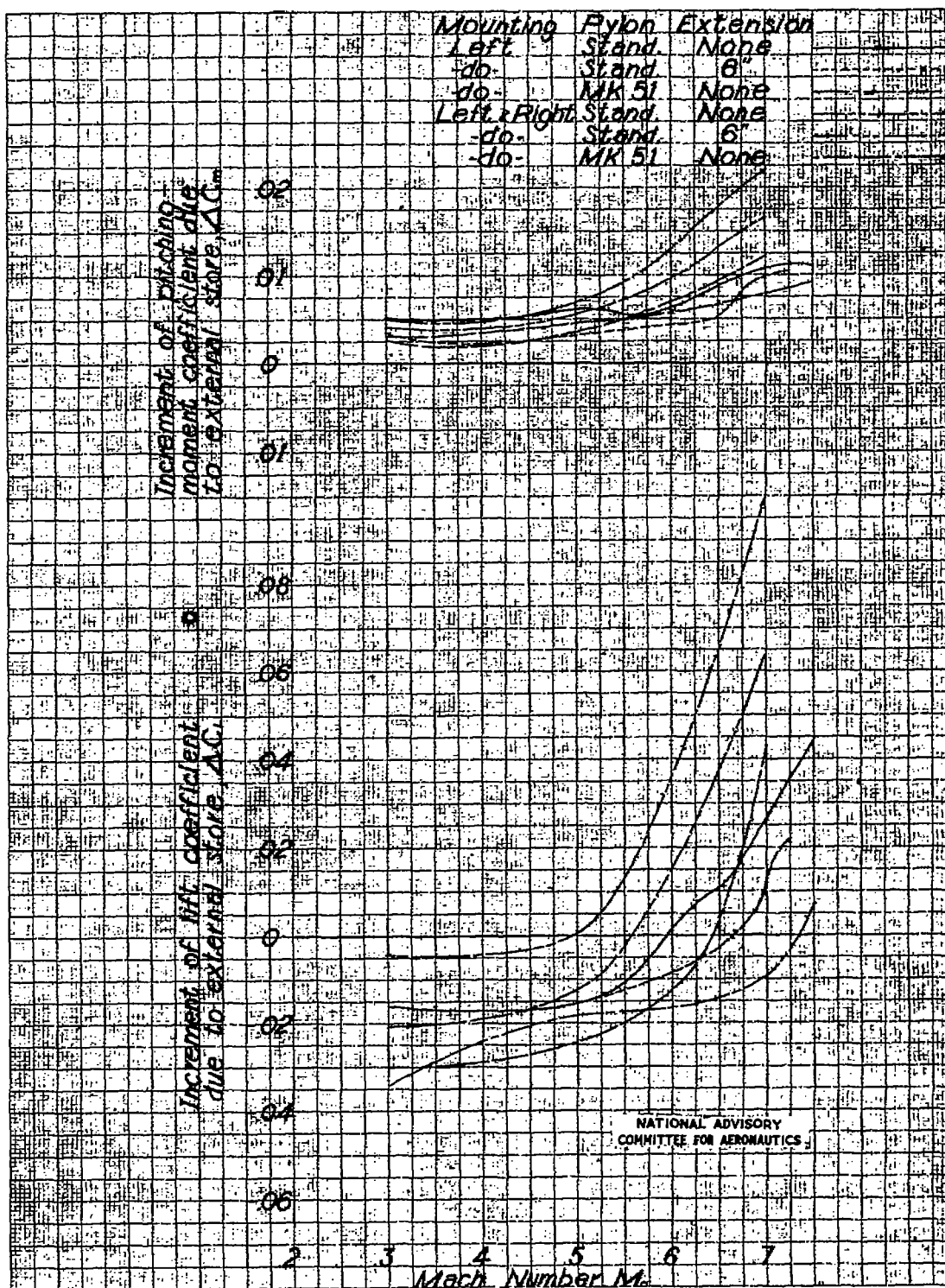drop tank.

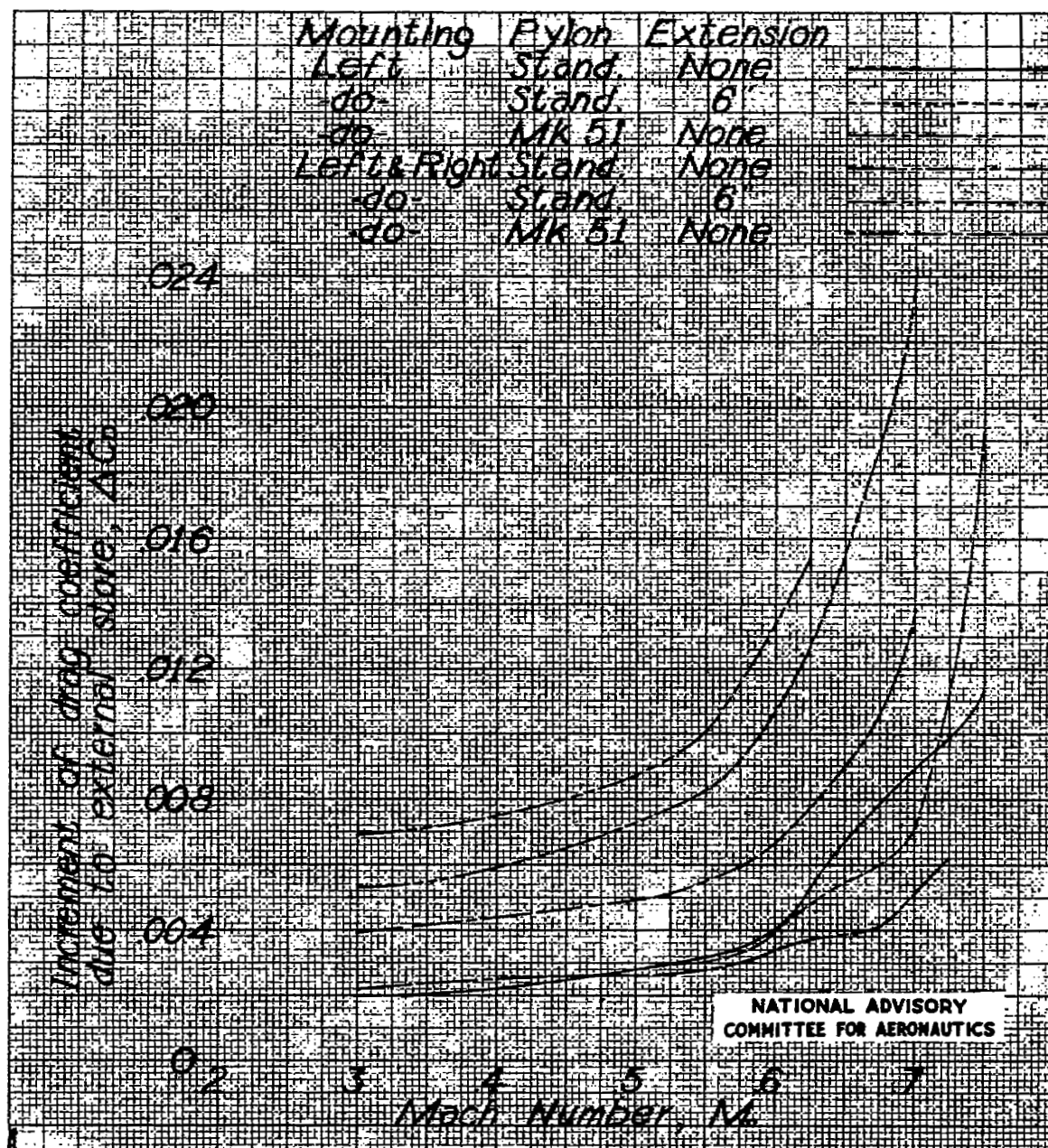(a) Concluded.

Figure 13.- Continued.

Mounting	Pylon	Extension
Left	Stand.	None
-do-	Stand.	6'
-do-	MK 51	None
Left & Right	Stand.	None
-do-	Stand.	6'
-do-	MK 51	None

Increment of drag coefficient due to external store, ΔC_D

Mach Number, M

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(b) $\alpha = -2.75^{\circ}$.

Figure 13.- Continued.

(b) Concluded.

Figure 13.- Continued.

Mounting	Pylon	Extension
Left	Stand.	None
-do-	Stand.	6'
-do-	Mk 51	None
Left & Right	Stand.	None
-do-	Stand.	6'
-do-	Mk 51	None

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(c) α = -1.5°.

Figure 13.- Continued.

(c) Concluded.

Figure 13.- Continued.

Mounting	Pylon	Extension
Left	Stand.	None
-do-	Stand.	6"
-do-	MK 51	None
Left+Right	Stand.	None
-do-	Stand.	6"
-do-	MK 51	None

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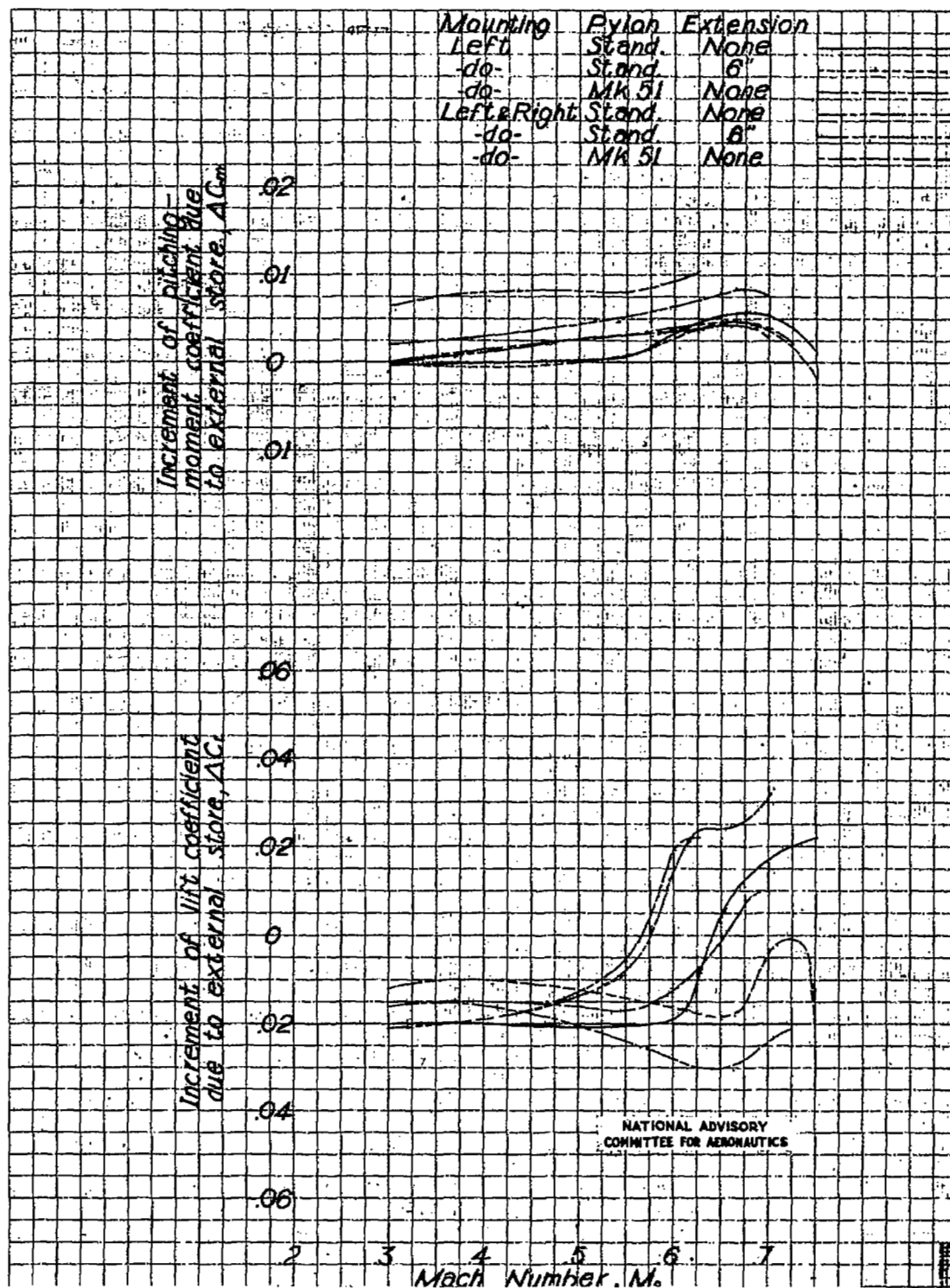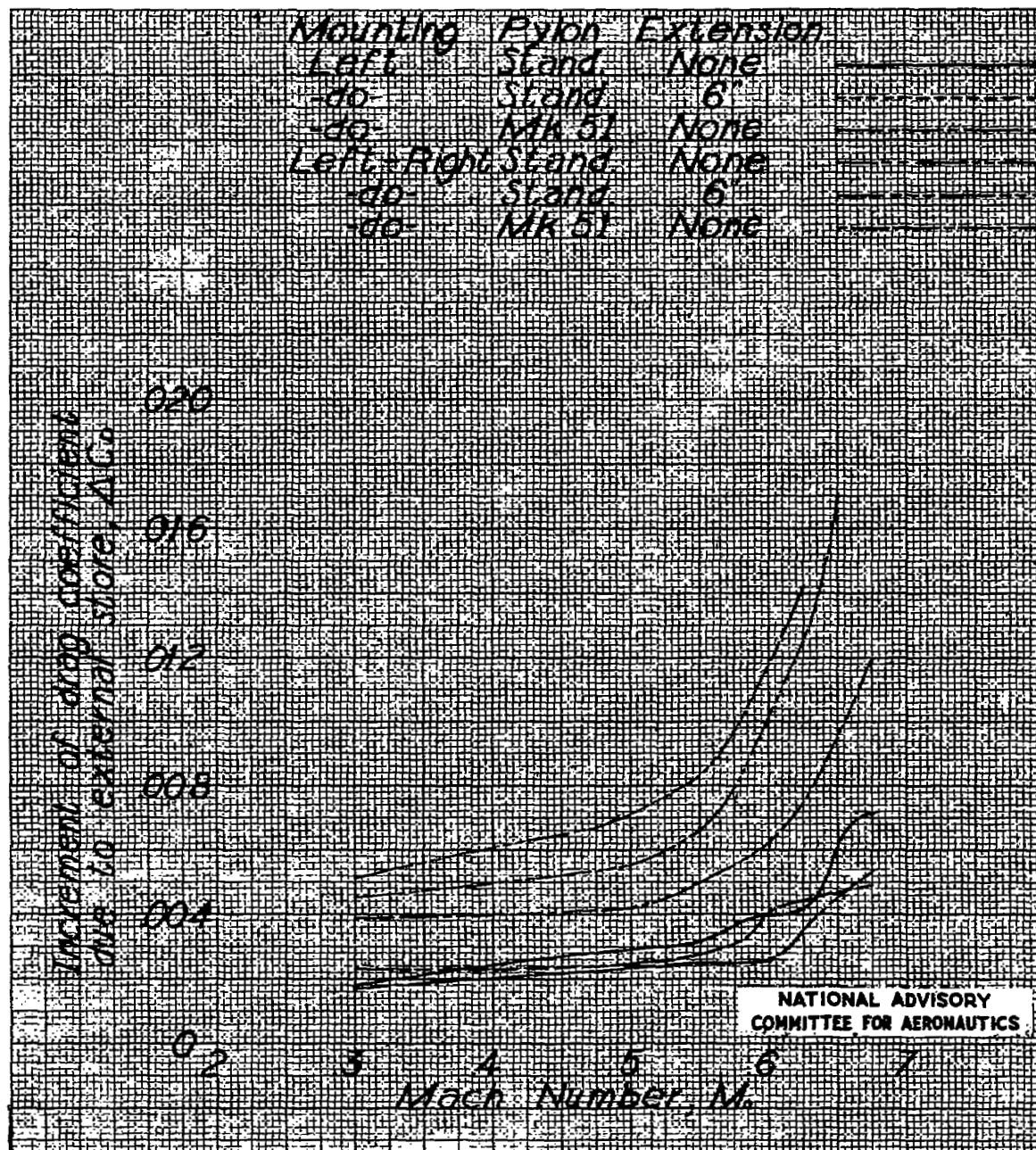
(d) $a = 0.25^{\circ}$.

Figure 13.- Continued.
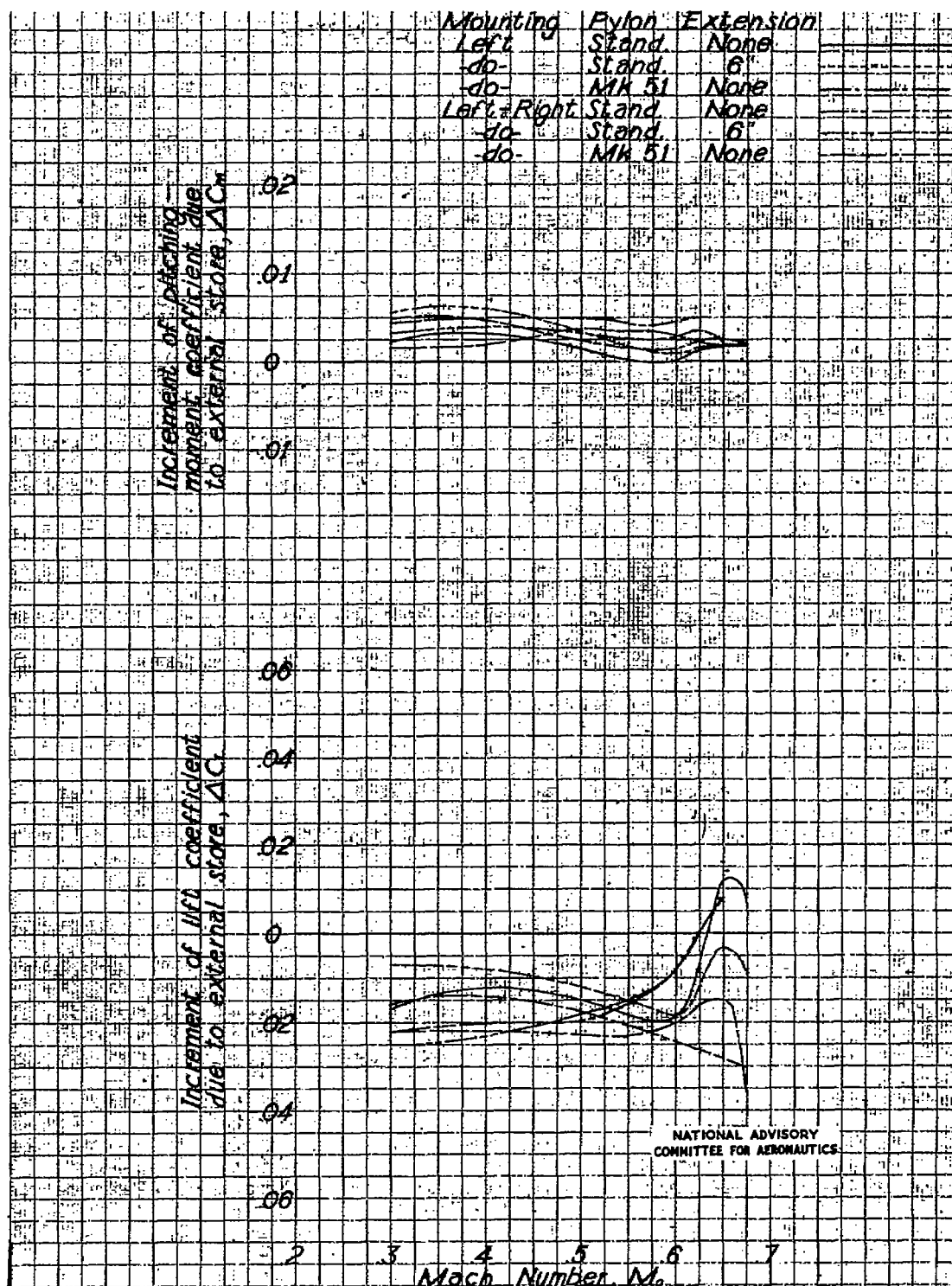
(d) Concluded.

Figure 13.- Concluded.

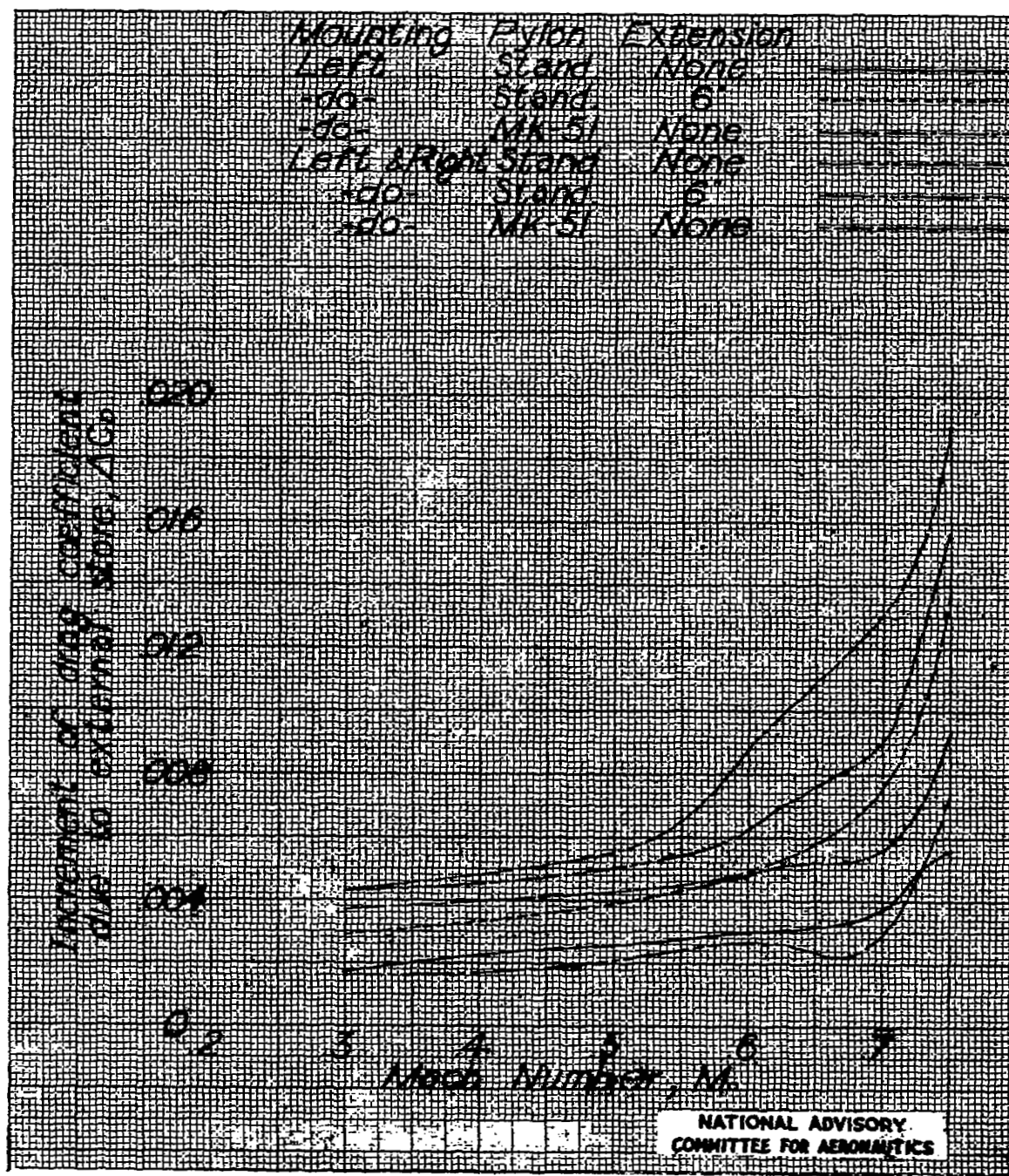| Mounting	Pylon	Extension
Left	Stand	None
-do-	Stand	6'
-do-	Mk 51	None
Left & Right	Stand	None
-do-	Stand	6'
-do-	Mk 51	None
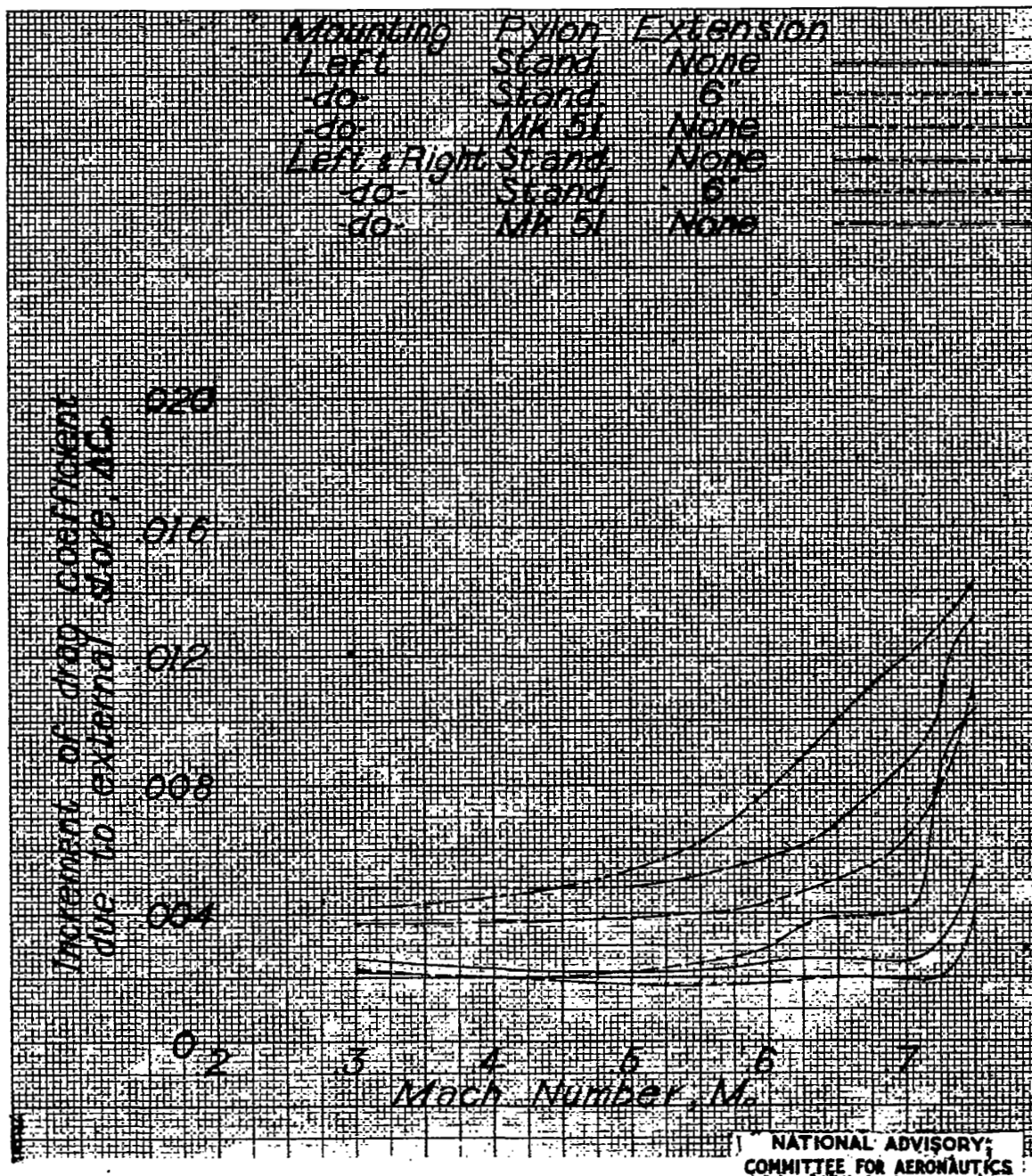
(a) α = -3.5°.

Figure 14.-   Effect of various pylon configurations on the aerodynamic
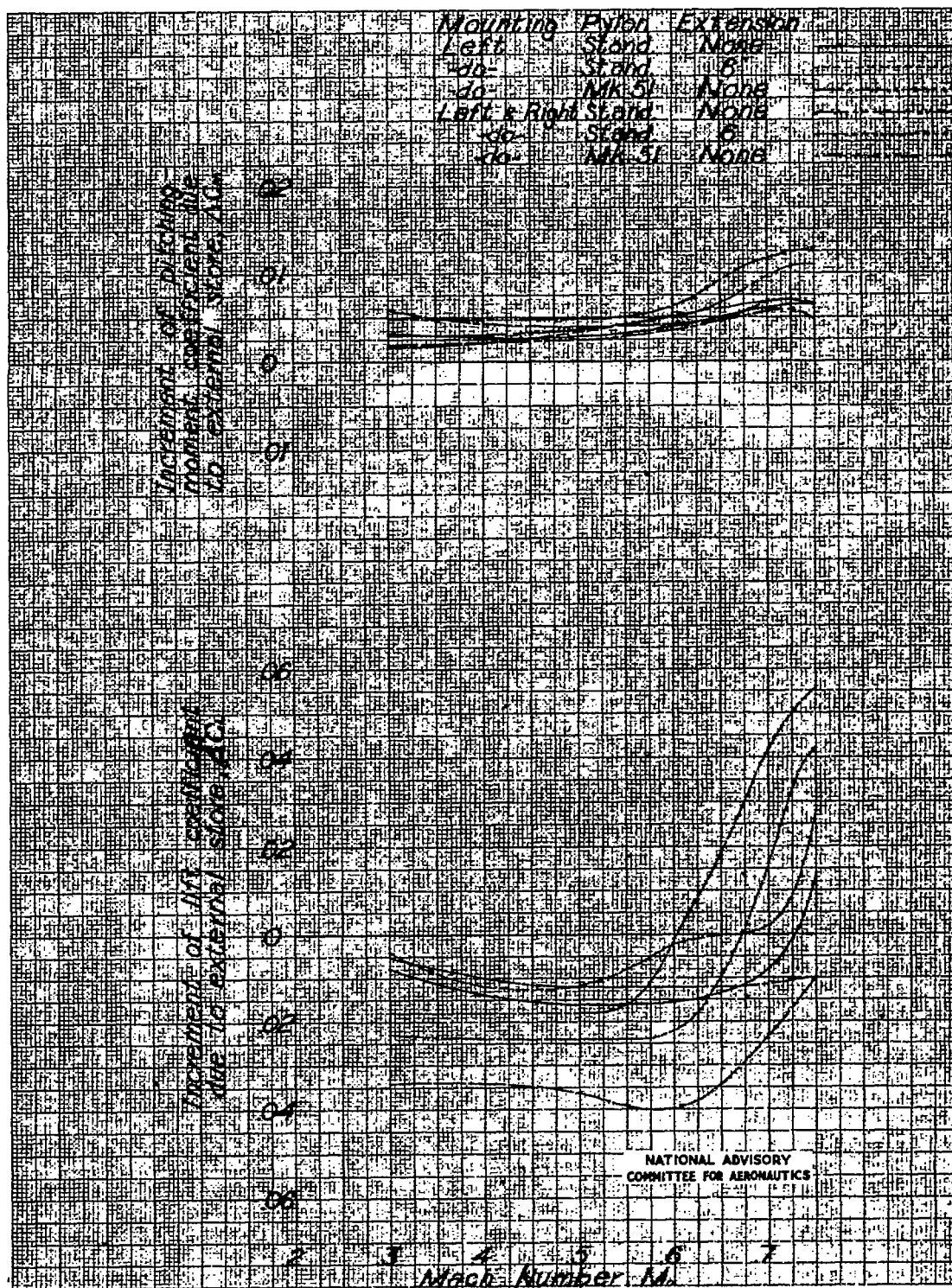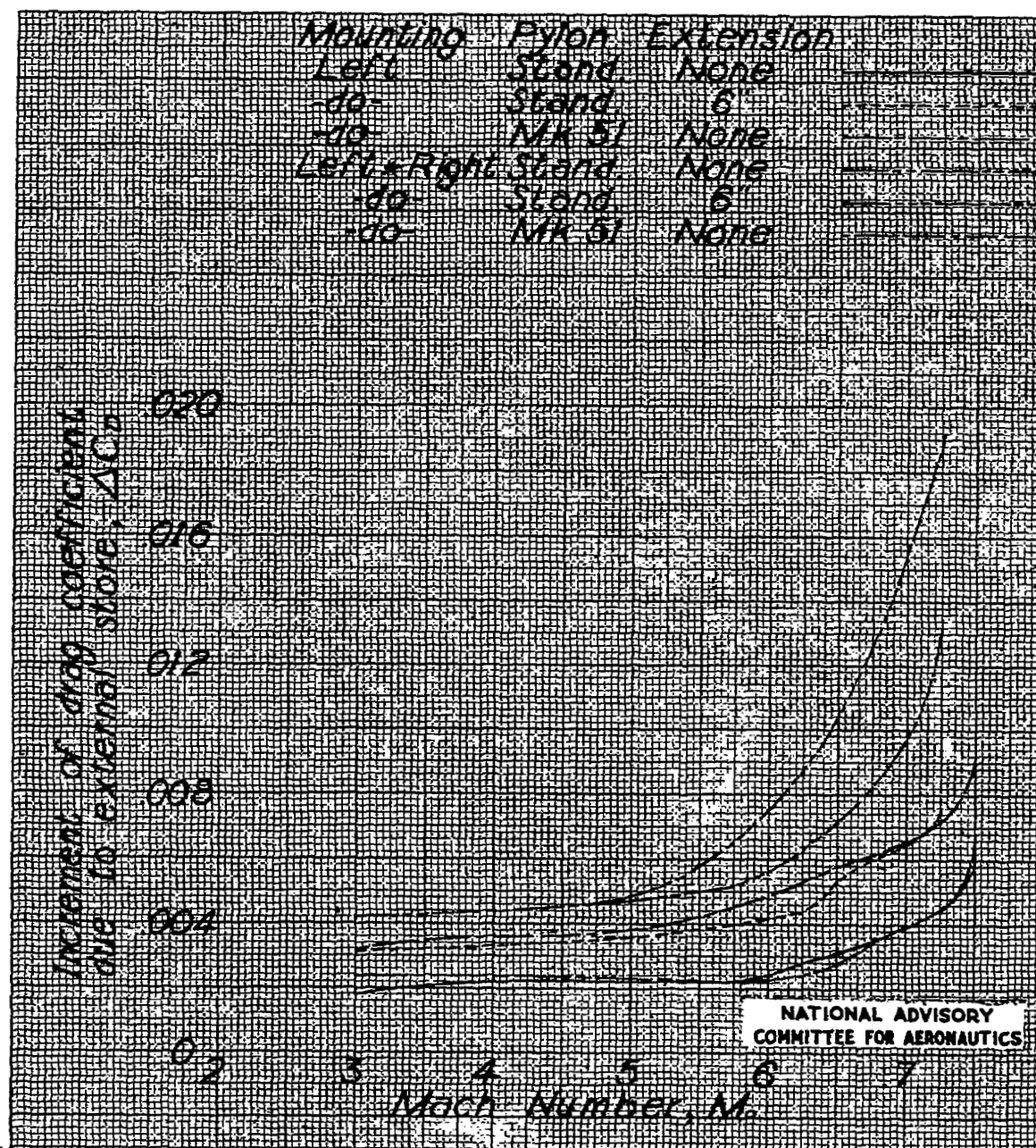characteristics of a model of a fighter-type airplane with Universal
drop tank.

Mounting	Pylon	Extension
Left	Stand	None
-do-	Stand	6"
-do-	Mk 51	None
Left & Right	Stand	None
-do-	Stand	6"
-do-	Mk 51	None

Increment of pitching moment coefficient due to external store, ΔC_m

.02

.01

0

-.01

Increment of lift coefficient due to external store, ΔC_L

.06

.04

.02

0

-.02

-.04

-.06

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.2   .3   .4   .5   .6   .7

Mach Number, M_0
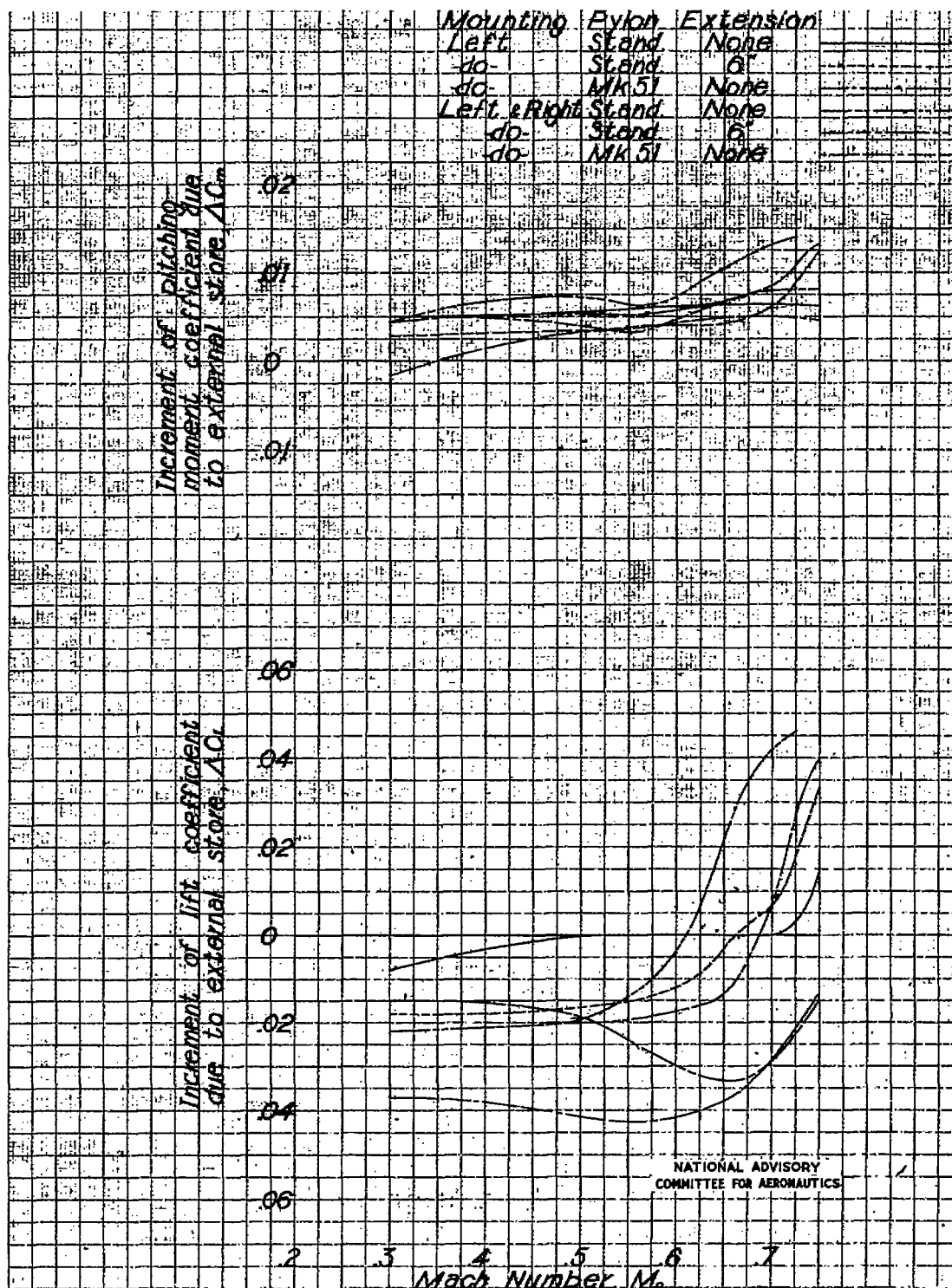
(a) Concluded.

Figure 14.- Continued.

(b) $\alpha \approx -2.75^{\circ}$.

Figure 14.- Continued.

(b) Concluded.

Figure 14.- Continued.

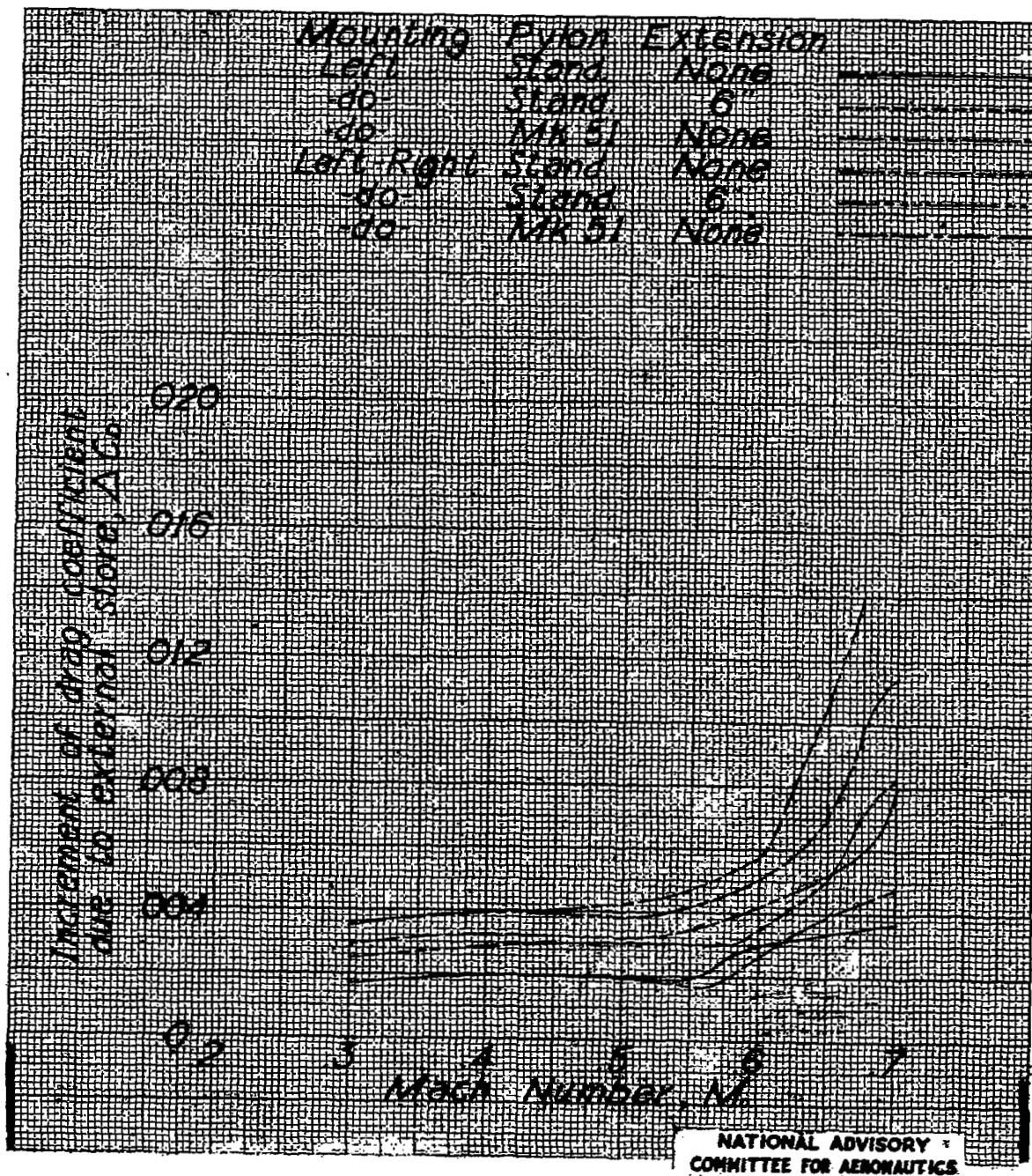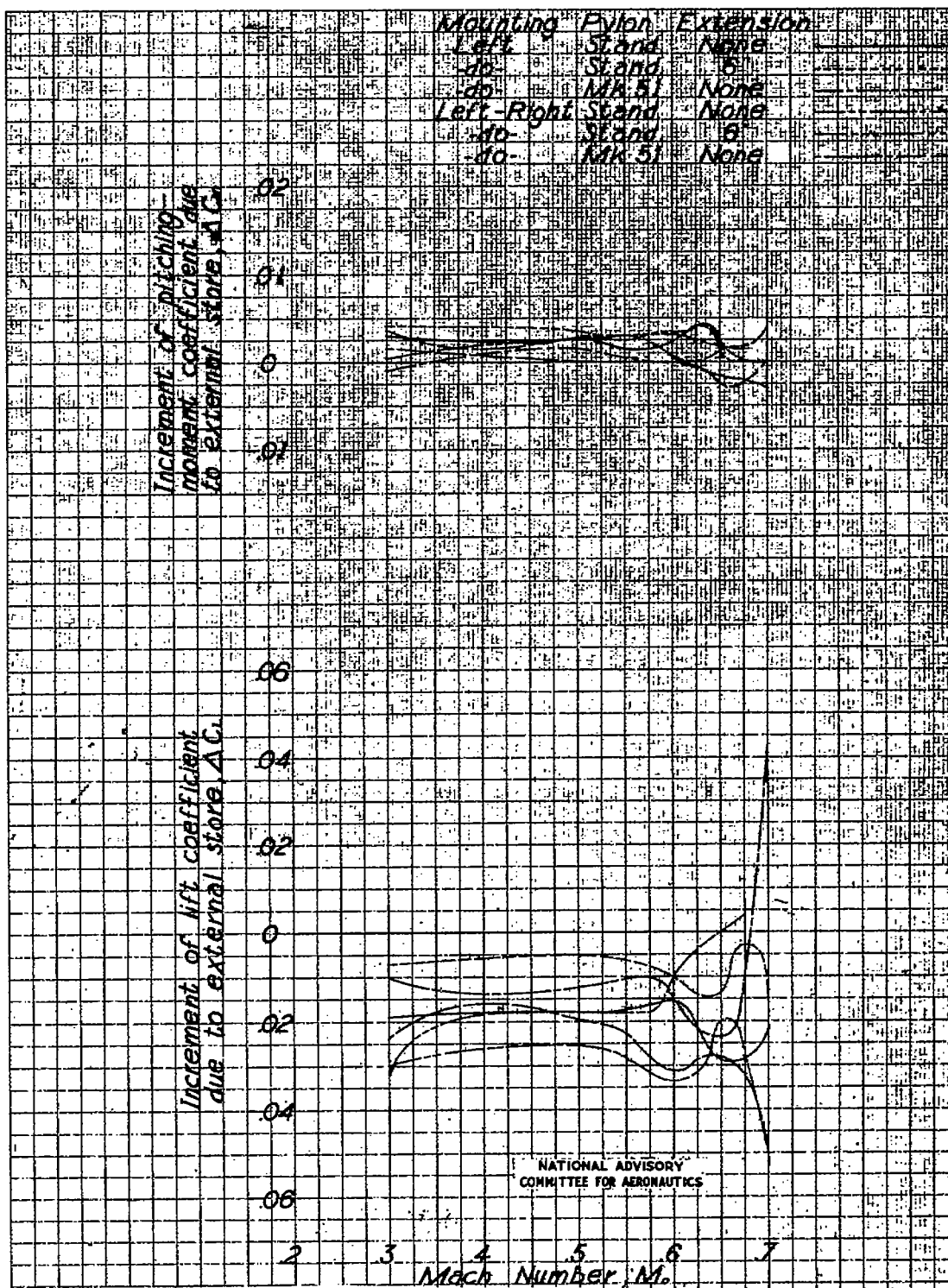| Mounting	Pylon	Extension
Left	Stand	None
-do-	Stand	6'
-do-	Mk 51	None
Left + Right	Stand	None
-do-	Stand	6'
-do-	Mk 51	None

(c) $\alpha = -1.5°$.

Figure 14.- Continued.
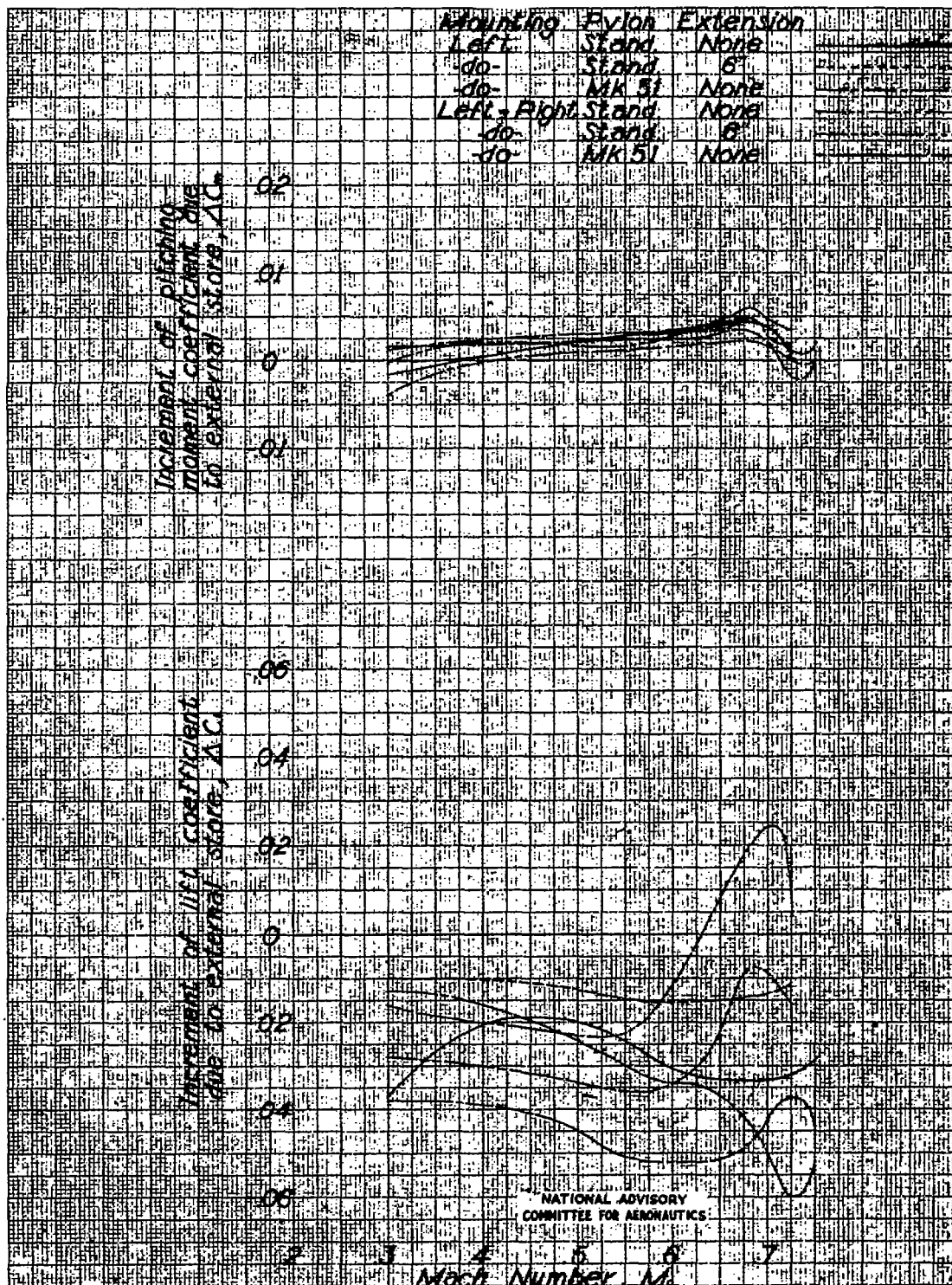
(c) Concluded.

Figure 14.- Continued.

(d) $\alpha = 0.25^{\circ}$.

Figure 14.- Continued

(d) Concluded.

Figure 14.- Concluded.